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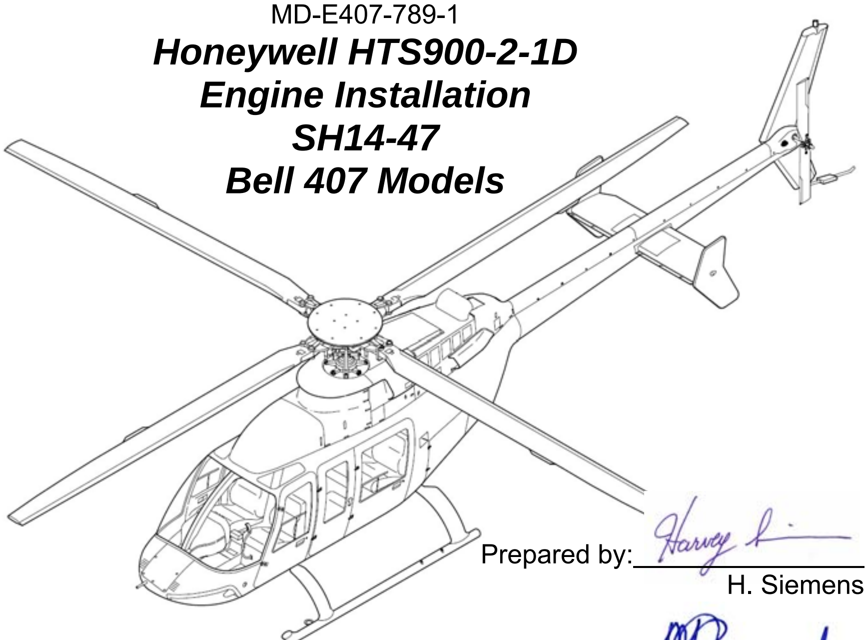
ROTORCRAFT MANUFACTURER'S DATA

MD-E407-789-1

Honeywell HTS900-2-1D Engine Installation SH14-47 Bell 407 Models



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Revision: **1**
Date: 05 JAN 2015

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Log of Revisions

Revision 0 .... 15 JUL 2014
Revision 1 ....05 JAN 2015

List of Effective Pages

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Intro-2	1	1-25	0	1-54	0
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Revisions

Revised text is indicated by a black vertical line. Insert latest revision pages; dispose of superseded pages.

General Information

This Manufacturer's Data is provided for use in conjunction with Flight Manual Supplement FMS-E407-789-1 for all Bell 407 modified per TCCA STC SH14-47/FAA STC SR03496NY. This manual contains useful information to familiarize the operator with the helicopter and its systems, to facilitate ground handling and servicing procedures, and to assist in flight planning and operations. All relevant information from the basic Bell 407 Manufacturer's Data has been incorporated into this document for the convenience of the operator. Therefore, there is no need to refer to the basic Bell 407 Manufacturer's Data.

If the sections or paragraphs have ivory background, it denotes that the information is original from the Bell 407 Manufacturer's Data.

If the section or paragraph is part of the amended information it has no special formatting.

This manufacturer's data is divided into three sections as follows:

Section 1 System Description
Section 2 Handling and Servicing
Section 3 Conversion Table
Section 4 Deleted

Section 1

System Description

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Section 1

System Description

1.1 Introduction

The Eagle 407HP helicopter, primary and auxiliary systems, and emergency equipment are described within this section. Optional equipment systems which do not require a Flight Manual Supplement (FMS) will be described in this section.

1.2 Helicopter Description

The Eagle 407HP is a single engine, seven-place light helicopter. Standard configuration provides for one pilot and six passengers.

The fuselage consists of three main sections: the Forward Section, the Intermediate Section, and the Tail boom Section. The forward section utilizes aluminum honeycomb and carbon graphite structure and provides the major load carrying elements of the forward cabin. The intermediate section is a semi-monocoque structure which uses bulkheads, longerons and carbon fibre composite side skins. The tailboom is an aluminum monocoque construction which transmits all stresses through its external skins.

The helicopter is powered by a Honeywell HTS900-2-1D engine. Refer to paragraph 1- 11, PowerPlant, for complete system description.

The main rotor is a four-bladed, soft-in-plane design with a composite hub and individually interchangeable blades. The tail rotor is a two-bladed teetering rotor that provides directional control. Refer to paragraph 1-28, Rotor System, for a complete system description.

Basic helicopter landing gear is the low skid type. Optional pop-out emergency flotation gear or high skid gear is also available.

1.3 Principal Dimensions

Principal exterior dimensions are shown in Figure 1-1. All height dimensions must be considered approximate due to variations in loading and landing gear deflection. Principal interior dimensions of the cabin and baggage compartment are shown in Figure 1-2.

1.4 Location References

Locations on and within the helicopter can be determined in relation to the fuselage stations, waterlines, and buttock lines measured in inches (mm) from known reference points.

1.4.A Fuselage Stations

Fuselage stations (FS or STA) are vertical planes perpendicular to, and measured along, the longitudinal axis of the helicopter. Station zero is the reference datum plane and is 1.0 inch (25.4 mm) forward of the nose of the helicopter or 55.16 inches (1401 mm) forward of the forward jack point center line.

1.4.B Waterlines

Waterlines (WL) are horizontal planes perpendicular to, and measured along, the vertical axis of the helicopter. Waterline zero is a reference plane located 20.0 inches (508 mm) below the lowest point on the fuselage.

1.4.C Buttock Lines

Buttock lines (BL) are vertical planes perpendicular to, and measured to the left and right, along the lateral axis of the helicopter. Buttock line zero is a plane at the lateral centerline of the helicopter.

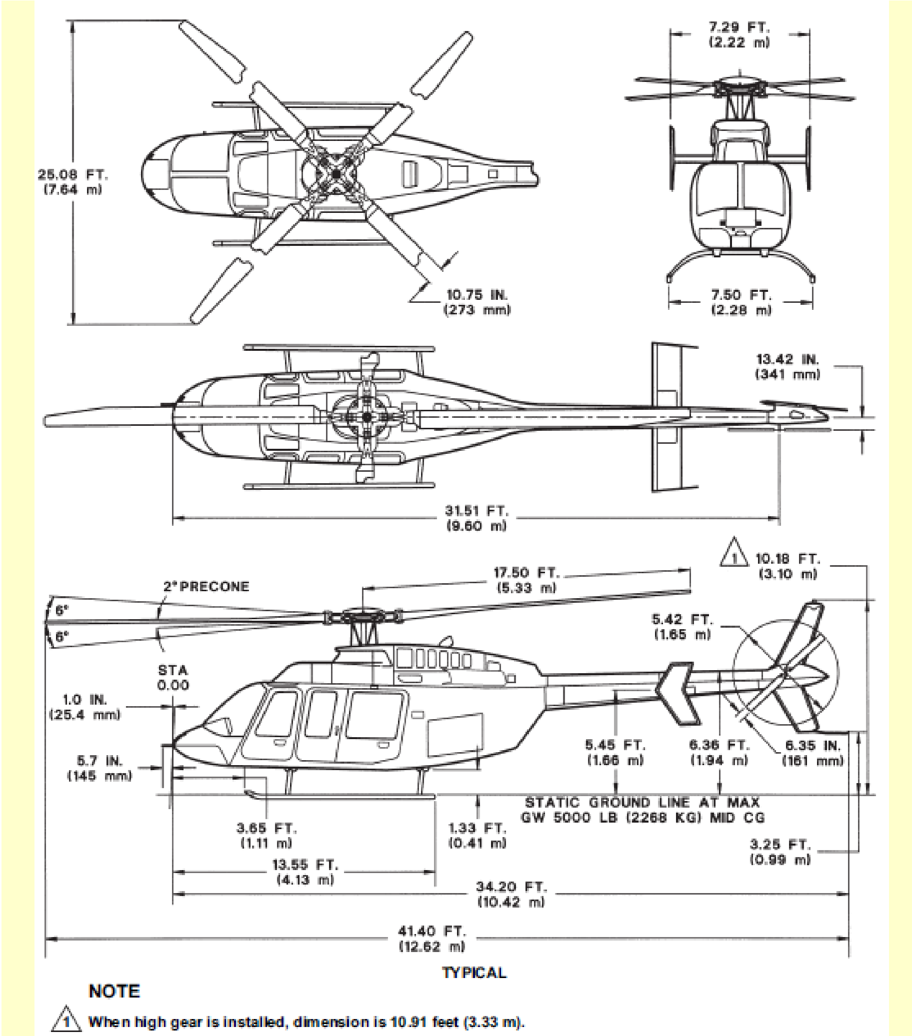
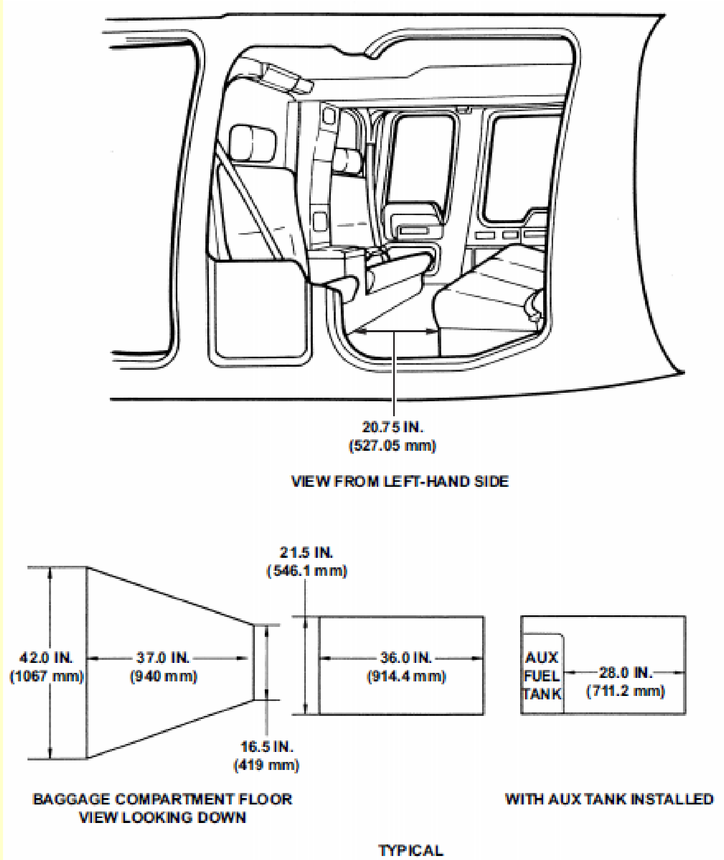


Figure 1-1 Principal Dimensions

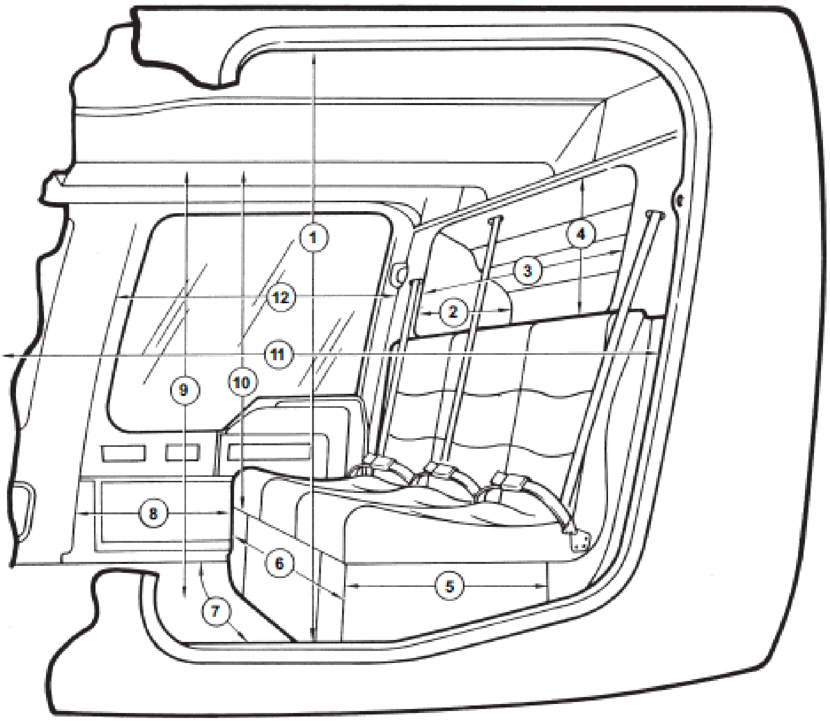


NOTES

1. Aft cabin space 85 cubic feet (2.4 m^3)
   Left forward cabin 20 cubic feet (0.6 m^3)
   Cabin floor stressed for 75 pounds per square foot (3.7 kg/100 cm^2).

2. Baggage compartment 16 cubic feet (0.45 m^3)
   Baggage compartment floor stressed for 86 pounds per square foot (4.2 kg/100 cm^2).

Figure 1-2 Aft Cabin and Baggage Compartment (Sheet 1 of 2)



TYPICAL
VIEW FROM LEFT-HAND SIDE

1. 39.3 inches (997 mm)
2. 11.0 inches (279 mm) deep
3. 36.5 inches (927 mm) forward side
4. 12.0 inches (305 mm) high forward side
5. 18.0 inches (457 mm)
6. 46.0 inches (1168 mm)
7. 49.0 inches (1245 mm)
8. 15.5 inches (394 mm)
9. 43.5 inches (1105 mm)
10. 38.0 inches (965 mm)
11. 58.5 inches (1486 mm) including litter door
12. 33.5 inches (851 mm)

Figure 1-2 Aft Cabin and Baggage Compartment (Sheet 2 of 2)

1.5 General Arrangement

The fuselage assembly (Figure 1-3) consists of three main sections: the forward section which extends from the cabin nose to the bulkhead aft of the passenger compartment, the intermediate section which extends from the bulkhead aft of the passenger compartment to the tailboom attachment bulkhead, and the tailboom section.

The forward section provides for pilot and passenger seating, the fuel cell enclosures, and the pylon support. The basic structure of the forward fuselage utilizes two honeycomb panels which are connected to create the floor, and an aluminum roof beam assembly which is attached to the center of another honeycomb panel to create the roof. Two bulkhead assemblies and a center post connect the floor and roof to make an integrated structure. The forward fuselage section is closed out by two carbon fiber composite side body fairings with interchangeable composite doors with automotive type latches.

The landing gear is attached to the bottom of the forward and aft bulkheads. The gear uses a three point attachment configuration to prevent ground resonance. The skid type landing gear consists of two skids attached to the ends of two arched crosstubes that are secured to the fuselage by means of a three point attachment configuration. Each skid tube is fitted with a tow fitting, two saddles with sockets for crosstubes, skid shoes along the bottom, a rear cap, and two eyebolt fittings for mounting of ground handling gear.

The intermediate section is a semi-monocoque structure which uses bulkheads, longerons, and carbon fibre composite side skins which do not need stiffeners or stringers. The lower section of the intermediate fuselage is closed out by a honeycomb composite fairing. The intermediate section provides a deck for engine installation, a baggage compartment, and an equipment compartment under the engine deck for electrical equipment, air conditioning equipment, and the tail rotor control servo actuator. The engine pan and forward and aft firewalls of the engine compartment are manufactured from titanium.

The tailboom is a monocoque construction which transmits all stresses through its external skins. To increase the strength of the structure, certain skins incorporate bonded doublers. The tailboom assembly includes the tail rotor driveshaft, pitch change mechanism, tail rotor gearbox, tail rotor hub and blade assembly, and the horizontal stabilizer and vertical fin.

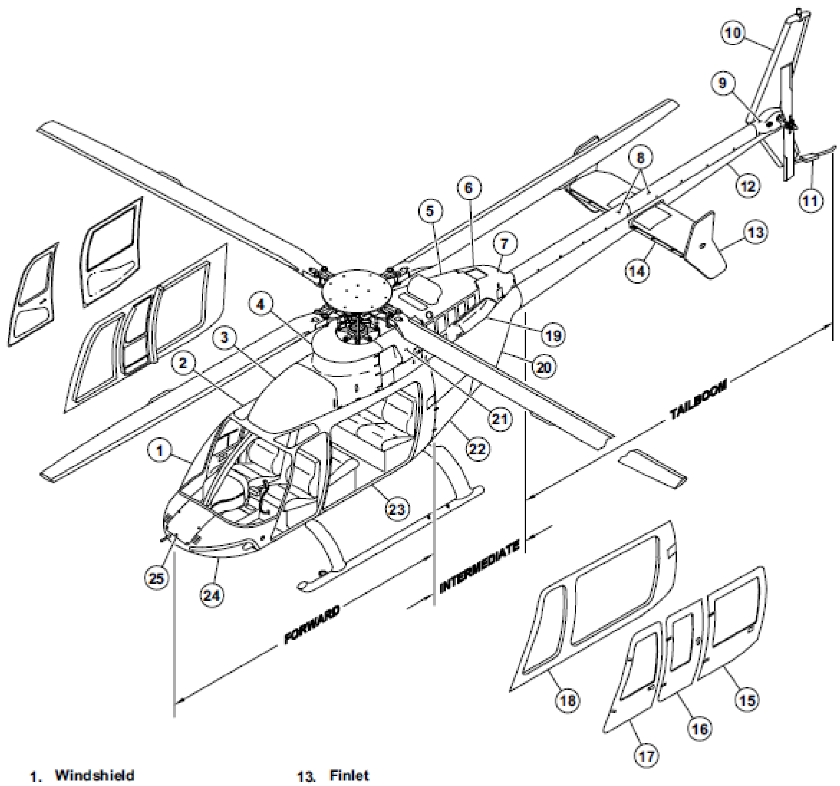
Cowlings and fairings which enclose the various roof and tailboom mounted assemblies provide inspection access via the use of hinge mounts, access doors, and inspection windows or cutouts. They are manufactured from either composite or aluminum materials and are readily removable for maintenance access.

The forward cowling is a composite construction and incorporates a single point hinge fitting which is located at the forward end of the fairing. Two flush mounted toggle hook latches secure the cowling. Support rods are internally stowed on each side of the cowl and fit into roof mounted clips to support the cowl in a raised position.

The transmission cowling is a composite construction and encloses the forward half of the main transmission. This cowling is mounted with fasteners. The fasteners are contained in metal ejector clips which ensure they are maintained with the cowling during removal, and which ensure easy identification of any fastener that may have dislodged from its receptacle.

The air induction cowling is an aluminum construction and encloses the aft half of the main transmission. Inlet ducts are provided on each side of the cowling to direct the airflow into the induction screen or particle separator. A small removable window is provided on either side of the cowling to allow checking of the inlet area when the particle separator is installed. An inspection cutout is provided on the right side to allow viewing of the transmission oil level sight gauge. Two hinged doors with flush-type latches are also provided for inspection of the aft transmission area

On the Eagle 407HP, a crossover duct is used to transfer inlet airflow from the existing inlet to the HTS-900-2-1D engine inlet, which opens upward. This crossover duct is concealed with a new fairing that sits on top pf the existing engine cowling



1. Windshield
2. Skylight window
3. Forward fairing
4. Transmission fairing assembly
5. Engine cowl
6. Aft fairing
7. Engine oil tank access door
8. Tail rotor driveshaft cover
9. Tail rotor gearbox fairing
10. Vertical fin
11. Tail skid and weight
12. Tailboom
13. Finlet
14. Horizontal stabilizer
15. Passenger door
16. Litter door
17. Crew door
18. Side body fairing
19. Oil cooler blower inlet duct
20. Aft skin panel
21. Air inlet cowl assembly
22. Baggage compartment
23. Forward fuselage
24. Lower window
25. Battery compartment door

Figure 1-3 Fuselage Assembly

The engine cowling has composite hinged side doors and an aluminum upper structure. The side doors incorporate flush-type latches and wing style stud fasteners. Attached to the side doors are oil cooler blower inlet ducts (S/N 53519 and subsequent or Post ASB 407-02-54). The doors are held in the open position with mechanical support devices. The side doors and upper structure of the engine cowling incorporate screened vents to allow air movement through the engine compartment.

The aft cowling is a composite construction and encloses the oil cooler and blower assembly and the engine oil tank. It incorporates a cutout to view the engine oil tank sight gauge, and two doors which are fastened with wing style stud fasteners. The cowling incorporates screened cooling vents.

The tailboom utilizes two composite constructed fairings to enclose the tail rotor driveshaft.

The tail rotor gearbox fairings are both composite constructed and secured with screws. The upper and lower fairings both incorporate hinge mounted doors for access.

1.5.A Crew Compartment

The crew compartment or cockpit occupies the forward part of the cabin (Figure 1-3). The pilot station is on the right side and the copilot/forward passenger station is on the left.

An instrument panel is mounted on a central pedestal in front of crew seats. The panel is tilted upward for maximum visibility from either seat.

The overhead console is centered on the forward cabin ceiling and incorporates most of the electrical systems circuit breakers and switches.

Each crew seat is covered with flame-retardant fabric and is equipped with a lap seat belt and a dual shoulder harness. Each shoulder harness contains an inertia re el which locks in the event of a rapid deceleration.

A door on either side permits direct access to the crew compartment. Each door is equipped with interior and exterior handles which operate a positive bolt latching mechanism. This style of door latching mechanism ensures smooth and positive operation when opening and closing the doors. Each exterior door handle incorporates a lock. The door windows are made of gray tinted acrylic plastic and incorporate a lower forward sliding window for ventilation.

The main windshields, lower cabin nose chin bubbles, and upper cabin roof skylight windows are also made of gray tinted acrylic plastic.

1.5.B Passenger Compartment

The aft area of the cabin (Figure 1-2) contains a space of 85 cubic feet (2.4 m^3) for the carrying of passengers or internal cargo. The cabin can be configured with utility, standard, or corporate interior kits. Each kit will include, but is not limited to, vacuum formed polycarbonate panels, armrests, door and sidewall magazine pockets, assist steps, door pulls, carpet, and all trim and attaching tracks and hardware.

Basic configuration includes two aft facing and three forward facing seats. All seats are covered with flame-retardant fabric and are equipped with lap seat belts and shoulder harnesses. The shoulder harnesses lock in the event of a rapid deceleration.

A cargo restraint kit is available for the installation of tie-down provisions. This kit provides forward bulkhead tie-down provisions using four shackle /eyebolt assemblies and floor mounted provisions using four anchor plates. These provisions allow cargo to be secured with a tie-down assembly. It is the responsibility of the pilot to ensure the cargo is adequately secured and uniformly distributed.

A door on either side permits direct access to the passenger compartment. In addition, the left passenger door is hinged on the litter door, so that the two may be opened together to aid in the loading of the helicopter. The LITTER DOOR caution light will illuminate if the litter door is not properly secured. Each door is equipped with interior and exterior handles, which operate a positive bolt latching mechanism. This style of door latching mechanism ensures smooth and positive operation when opening and closing the doors. Each exterior door handle incorporates a lock. All cabin windows are made of tinted acrylic plastic and the passenger doors can incorporate a lower forward sliding window for ventilation.

1.5.C Baggage Compartment

The baggage compartment (Figure 1-2) is located aft of the passenger compartment and has a capacity of 16 cubic feet (0.45 m^3). The compartment can carry up to 250 pounds (113.4 kg) of baggage or other cargo which can be secured using a tie-down assembly and the tie-down fittings provided. It is the responsibility of the pilot to ensure the cargo is adequately secured and uniformly distributed.

Access to the compartment is provided by an exterior door on the left side of the aft fuselage. The door is hinged at the forward end and opens the full width and height of the compartment. It is secured by two push-button latches and a keyed lock. The BAGGAGE DOOR caution light will illuminate if the door is not properly secured

A 19.2 US gallon (72.7 L) fuel tank may also be installed in the baggage compartment as an optional kit (BHT-407-FMS-6).

1.5.D Tailboom

The tailboom consists of the tailboom and the components it supports: tail rotor and drive system, vertical fin, and horizontal stabilizer (Figure 1-3).

The tail rotor drive system consists of a driveshaft mounted along the top of the tailboom and a gearbox, which reduces the tail rotor driveshaft input speed from 6317 to 2500 RPM. The gearbox also changes the direction of drive 90° to accommodate the tail rotor for directional control.

The vertical fin, composed primarily of aluminum and honeycomb construction, provides directional (yaw) stability and is mounted on the aft end of the tailboom on the right side. It contains a top fairing to mount the anticollision light and on the lower edge, a rubber bumper, and tail skid to protect the tail rotor and fin in the event of a tail low landing. The fin sweeps back, both above and below the tailboom. The leading edge is canted outboard 9° to reduce the required amount of tail rotor thrust during forward flight at cruise speed.

The horizontal stabilizer extends through the tailboom, has leading edge slats, and small auxiliary vertical fins or “dynamic dihedrals”. The main body of the horizontal stabilizer is a one-piece aluminum honeycomb structure. It is an inverted airfoil which provides a downward resultant lift on the tailboom to maintain the cabin in a nearly level attitude throughout all cruise airspeeds and to aerodynamically streamline the fuselage to reduce drag. The leading edge slats are designed to improve pitch stability during climbs.

The small auxiliary vertical fins are located on each end of the horizontal stabilizer. The leading edges of the fins are both offset 5° outboard of the helicopter centerline. This improves the roll stability of the helicopter in forward flight.

The tailboom utilizes two composite constructed fairings to enclose the tail rotor driveshaft. The cowlings are mounted with studs and receptacles. The studs are contained in metal ejector clips, which ensure the studs are maintained with the cowlings during removal, and which ensure the easy identification of any stud that may have dislodged from its receptacle.

The tailrotor gearbox fairings are both composite constructed and secured with screws. A hinged door with wing style stud fasteners is incorporated on the top fairing for ease of servicing the gearbox. A hinged door is incorporated on the bottom of the fairing for ease of access to the tail rotor gearbox chip detector. The tail rotor gearbox oil level may be viewed through the aft screened cooling vent of the fairing.

1.6 Instrument Panel, Console and Pedestal

1.6.A Instrument Panel

Figure 1-4 shows the instrument panel with optional equipment installed. The instrument panel provides vibration dampening and is hinge mounted on a central pedestal in front of the crew seats.

The instrument panel is tilted at a 5° angle for maximum visibility. The flight instruments are located on the right side of the panel, and the systems instruments are in two rows to the left of the flight instruments. The caution and warning panel is mounted just below the glareshield across the top of the instrument panel.

1.6.B Overhead Console and Pedestal

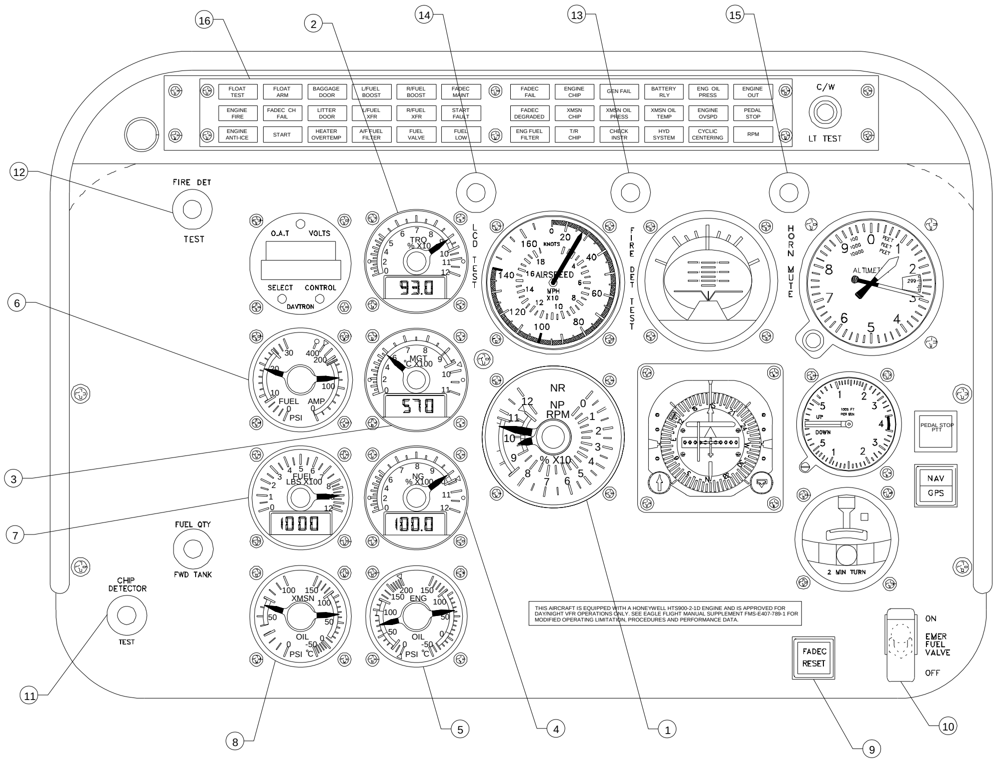
The overhead console (Figure 1-5) is centered on the cabin ceiling and incorporates most of the electrical systems' circuit breakers and switches. The forward section of the panel has integral lighting controlled by the instrument lighting rheostat knob.

The pedestal (Figure 1-4) extends from the instrument panel downward and aft to the cabin seat structure. This forms a mounting platform for optional equipment such as the audio panel, radios, and gyrocompass control panel etc. The lower left side of the pedestal (Figure 1-6) contains the FADEC/ECU ground interface port, ENGINE INSTRUMENT maintenance port, 28 VDC auxiliary receptacle, and the M/R RADS maintenance port and GPS data loader port.

1.7 Engine Instruments

The propulsion instruments are digital indicators with digital and analog representation that are individually powered through circuit breakers located on the 28 VDC bus of the helicopter. Engine instruments consist of TORQUE, MGT, NG, NP/NR and Engine Oil Press & Temp indicators.

With the exception of the Engine Oil Press & Temp Indicator and the Dual Tachometer Indicator, the instruments consist of an analog pointer with LCD digital display. The Dual Tachometer Indicator and Engine Oil Press & Temp Indicator consist of analog pointers only.



1. Dual Tachometer Indicator
2. Torque Indicator
3. MGT Indicator
4. NG Indicator
5. Engine Oil Press & Temp Indicator
6. Fuel Press & Ammeter Indicator
7. Fuel Quantity Indicator
8. Transmission Oil Press & Temp Indicator
9. FADEC Reset Switch
10. Emergency Fuel Valve Switch
11. Chip Detector Test Switch
12. Fire Detect Test Switch (Alternate Location)
13. Fire Detect Test Switch
14. LCD Test Switch
15. Horn Mute Switch
16. Caution/Warning Annunciator Panel

Figure 1-4 Instrument Panel and Pedestal (Sheet 1 of 2)

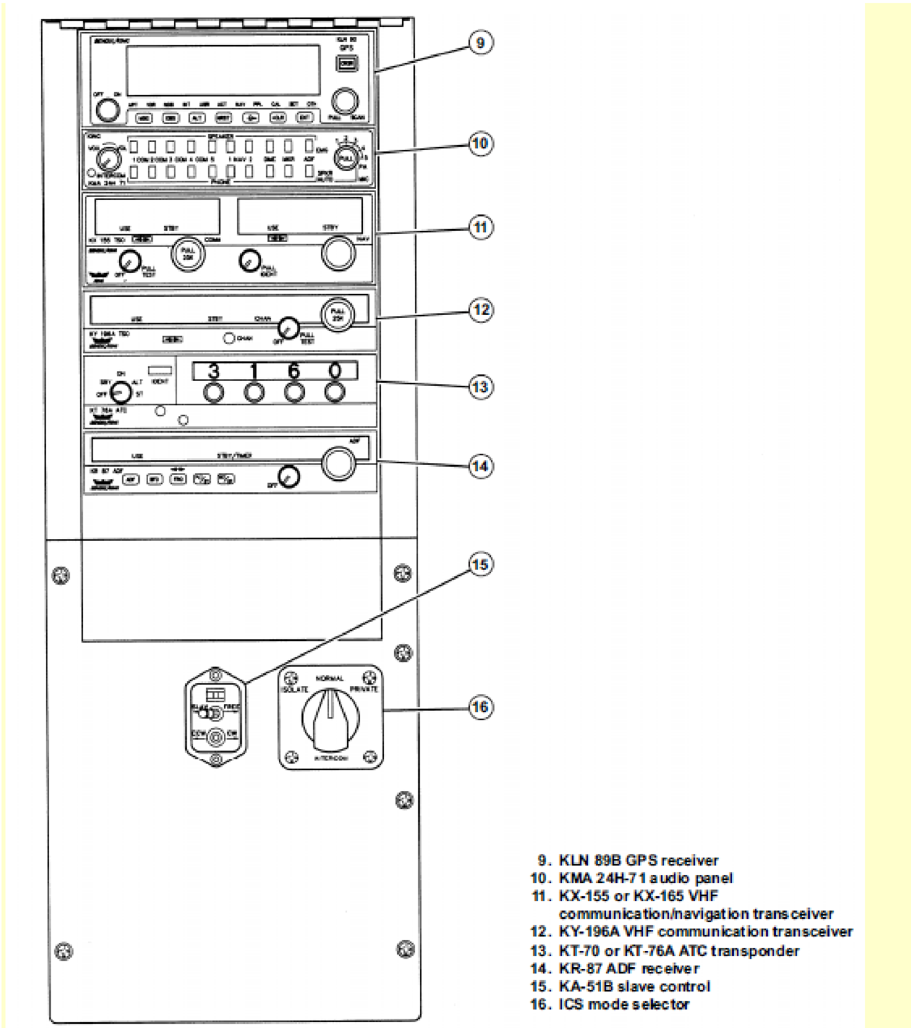


Figure 1-4 Instrument Panel and Pedestal (Sheet 2 of 2)

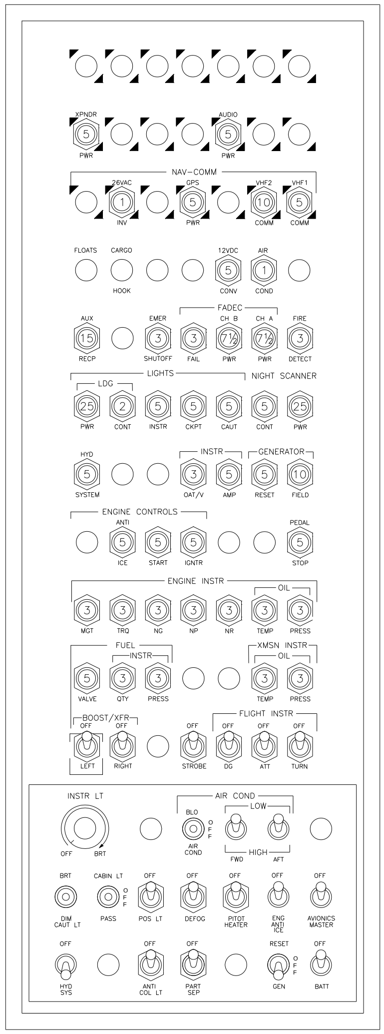


Figure 1-5 Overhead Console

Figure 1-6 Maintenance Port
(Left Lower Side Console)

1.7.A Instrument Operation

With the exception of the Engine Oil Temp & Press Indicator, all engine instruments receive ARINC 429 data from both FADEC/ECU channels for indication. Indicators monitor the active channel via bit 23 and 24 of word label 270. All of the propulsion instruments have a built-in-test (BIT) capability, where the indicators check integrity of internal components that causes the discrete signal output to come on momentarily during power-up and built-in test.

1.7.A.1 Power –On BIT

The power-on BIT starts when the power is applied to the instrument. The power-on BIT does an integrity check of the electronic components of the indicator.

NOTE

> Built-in test on the indicators causes discrete output to be tested and momentarily activates specific annunciator during power-on.

During the power-on BIT, all indicators with digits display the following sequence:

- All segments are lit providing an “8.8.8.8” pattern to verify that all display elements are functioning.
- The first three digits and then the last three digits of the software version are displayed for one second each.
- The Operation label “O-” is displayed if no over-torque event has occurred since the last time the indicator memory was reset. Otherwise, the Operation Exceedance label “OE” is displayed.
- If an exceedance has been recorded, the date of the highest engine parameter during operation since the last memory reset is displayed, followed by the time.
- The highest engine parameter over the pre-determined limit during operation since the last memory reset is displayed, followed by the

time.

- During this time, the indicator pointer indicates the engine current parameter.
- If the current Ng > 5%, the indicator will exit the playback mode and the digital display will start displaying the current parameter.
- Otherwise, the indicator will playback the Operation Exceedance Count label “OEC” followed by the number of exceedance events recorded during operation.

After the power-on display sequence, normal operation of the digits begins.

1.7.A.2 LCD Test

The LCD test starts when the LCD TEST button on the instrument panel is pushed. During the LCD test, the instruments with digital display playback portions of the recorded events. Indicators will display peak value of the exceedances recorded and “OEC” followed by a number. To acknowledge exceedance on any of the indicators and when CHECK INSTR light is activated by the exceedance, LCD TEST switch should be cycled to allow succeeding exceedance event. On the Dual Tachometer indicator, the individual Rotor and Turbine pointers are driven to indicate 100% during LCD test when Ng < 5%.

1.7.A.3 Exceedance Monitoring

TORQUE, MGT, NG and NP/NR instruments have the ability to record engine exceedances. Exceedances have been predefined at the design stage and preprogrammed into the microprocessor in each indicator. Exceedances are defined as limits of operation above which there may be some maintenance action required.

NOTE

Exceedance monitoring is provided as an aid to determining required maintenance action. Only the person in command of the helicopter at the time the exceedance occurs can verify that the exceedance recorded reflects the actual occurrence.

The above instruments have built in NVM that allows them to store up to 63 exceedance events. For each exceedance, the memory records the date, duration, and peak value during the exceedance. If exceedance events recorded are greater than 63, the earliest dated exceedance that was in memory is deleted and the latest exceedance is added in its place.

To provide notice to the pilot that an exceedance is recorded, these indicators also have other preprogrammed advisory points at which they will flash the digital display. Specific values are discussed in applicable systems descriptions in this section of manufacturer's data.

When a potential exceedance is detected by the instrument, it will start flashing the digital display and stop flashing if the pilot makes control input to reduce the instrument readings below the advisory values. When the indicator exceeds specifically preprogrammed values, an exceedance will be recorded. When an exceedance is recorded, the CHECK INSTR light will be turned on by the instrument.

The pilot can acknowledge the exceedance and cause the peak value to be displayed on the digital display by pushing the LCD TEST button (Figure 1-4) on the instrument panel. If there have been exceedances recorded on different indicators, each indicator will display its last exceedance.

The last exceedance will continue to display each time the indicator is powered up until the exceedance(s) is removed from the NVM of the indicator using a computer with maintenance download software (See ICA-E407-789)

Exceedances can be downloaded from the indicators through the Instruments Maintenance Port RS485. Refer to Figure 1-6. Exceedances are also monitored and recorded by the FADEC/ECU. For determining the maintenance requirements, always use the exceedances recorded by the instruments in conjunction with the Honeywell LMM for HTS900-2-1D.

1.7.B Last Flight Cycle Count

The MGT Indicator and NG Indicator are programmed to display the last flight power turbine cycle count and last flight gas producer cycle count respectively when Ng < 5%. ARINC 429 data is transmitted by the FADEC ECU to the indicators for processing, while the indicators saves the data on the non-volatile memory so that it is available to download for maintenance purposes.

1.8 Check Instrument (CHECK INSTR) Light

The CHECK INSTR light circuit is designed to alert the pilot that either the TORQUE, MGT, NP/NR, NG, Engine Oil Temp & Press or Fuel Pressure and Ammeter Indicator has detected an exceedance or the Fuel Quantity gauge detected fault on the fuel conditioner.

When any of these indicators exceed preset values, the CHECK INSTR light will turn on. The digital readout display will also flash at predefined limits. The CHECK INSTR light will remain on until the pilot acknowledges the exceedance by pushing the LCD TEST button (Figure 1-4).

1.9 Clock

The clock is included in a multifunction indicator mounted in the upper left area of the instrument panel. The indicator also displays Outside Air Temperature (OAT) in °C or °F and volts (DC). The clock is a digital display, quartz crystal chronometer.

The clock has a high contrast liquid crystal display and a two button control system designed to prevent accidental time setting while selecting various functions. The clock functions are as follows:

- Universal Time (UT) in 24-hour format
- Local Time (LT) in 12-hour format (24-hour format may be selected).
- Elapsed Time (ET) count up timer to maximum of 99 hours, 59 minutes.
- Elapsed Time (ET) count down timer from maximum 59 hours, 59 minutes.

Elapsed Time (ET) alarm(flashing display) at zero count down.
Flight Time (FT) count up to a maximum of 99 hours, 59 minutes.
Flight Time (FT) alarm (flashing display) at zero count down.

A line on the display will be positioned over the letters indicating the function selected (UT/LT/FT or ET).

The clock display receives its power from the 28 VDC bus through the OAT/V INSTR circuit breaker. When the clock is not powered, the CONTROL and SELECT buttons are disabled. The display lighting is powered by 5 VDC instrument lighting system.

When all electrical power is turned off, the clock display disappears, but the crystal timing reference continues to operate from the power of a 1.5 volt penlight dry cell clipped to the back of the clock case. The dry cell is not charged by the helicopter electrical systems, and it should be replaced annually to ensure uninterrupted operation of the timing reference.

1.9.A Clock Operation

The SELECT button selects what is to be displayed and the CONTROL button controls what is displayed. The SELECT button sequentially selects UT, LT, FT, ET and back to UT. The CONTROL button starts and resets the elapsed time when momentarily pressed.

The Flight Time is recorded based on 28 VDC power input received when the helicopter hourmeter is running (paragraph 1.39.B).

1.9.A.1 Universal Time Setting

Select UT using the SELECT button. Press the SELECT and CONTROL buttons simultaneously to enter the set mode. The tens of hours digit will start flashing. The CONTROL button controls the flashing digit and each push of the button increments the digit. Once the flashing digit is set, the SELECT button selects the next digit to be set, from left to right across the display. After the last digit is set, a final press of the SELECT button will exit the clock from the set mode. The function indicator will resume normal flashing to indicate the clock is running.

1.9.A.2 Local Time Setting

Select LT using the SELECT button. Press the SELECT and CONTROL buttons simultaneously to enter the set mode. The tens of hours digit will start flashing. The setting operation is the same as UT, except that the minutes are already synchronized with UT and cannot be set in LT.

1.9.A.3 Elapsed Time Count Up

Select ET using the SELECT button. Pressing the CONTROL button will start the ET counting up in minutes and seconds until reaching 59 minutes and 59 seconds. The ET will then start counting in hours and minutes up to 99 hours and 59 minutes. Pressing the CONTROL button again will reset the ET to zero.

1.9.A.4 Elapsed Time Count Down

Select ET using the SELECT button. Press the SELECT and CONTROL buttons simultaneously to enter the set mode. The count down time can now be set. Entering the time is identical to UT time setting. When the time is entered and the last digit is no longer flashing, the clock is ready to start the count down. Momentarily pressing the CONTROL button starts the count down. When the countdown reaches zero, the display will flash and the ET counter will begin counting up.

Pressing either the SELECT or CONTROL button will deactivate the flashing alarm. Pressing the SELECT button will select UT display or pressing the CONTROL button will reset the ET counter to zero.

1.9.A.5 Flight Time Reset

Select FT using the SELECT button. Press and hold down the CONTROL button for 3 seconds or until the display shows 99:59. Flight time will be zeroed upon release of the CONTROL button.

1.9.A.6 Flight Time Alarm Set

Select FT using the SELECT button. Press the SELECT and CONTROL buttons simultaneously to enter the set mode. The countdown time can now be set. Entering the time is identical to UT time setting. When the time is entered and the last digit is no longer flashing, the clock is ready to start the count down. The Flight Time setting has not been affected by setting the Flight Time Alarm. When the Flight Time begins recording in conjunction with the hourmeter (paragraph 1.39.B), the Flight Time alarm will begin its countdown.

1.9.A.7 Flight Time Alarm Display

NOTE

When the Flight Time is equal to the alarm time, the display will flash. If FT is not being displayed at the time the alarm becomes active, the clock will automatically select FT for display.

Press either the SELECT or CONTROL button to turn off the alarm. Flight time will remain unchanged and will continue counting.

1.9.A.8 Test Mode

Hold the SELECT button in for 3 seconds and the display will read 88:88. This indicates all digits are functioning properly.

1.10 Caution and Warning System

The main function of the caution and warning system is to detect specified conditions and provide a visual or visual/audio indication. Detection is accomplished through the use of monitoring circuits that, when actuated, cause the annunciator lights to illuminate. Visual indications are provided through a caution and warning panel while audio indications are provided through three separate warning horns: ENGINE OUT (pulsating), LOW ROTOR (continuous), and FADEC FAIL (chime tone) (Table 1-1).

The caution and warning panel comprises 36 individual annunciator positions which provide a visual indication of cautions (amber lights), warnings (red lights), and advisory (green or white lights) conditions. Each annunciator contains three lamps. Individual lamp operation is provided in the event one or more of the lamps burns out.

Of the 36 annunciators, 32 are illuminated with a ground input (ground seeking), three are illuminated with a 28 VDC bus input (positive seeking), and one is illuminated with a combination ground and 28 VDC bus input (ground/positive seeking)

The FLOAT TEST, ENG OIL PRESS, ENGINE OVSPD, ENGINE OUT, and RPM cannot be dimmed. The remaining lights may be illuminated in either the fixed bright or the fixed dim mode. With the instrument light rheostat knob in the OFF position, all lights will illuminate in the bright mode if they are turned on. With the instrument light rheostat knob, set between the dim and BRT position, the CAUT LT DIM/BRT switch may be used to select either the dim or bright mode. In the DIM mode all lights, with the exception of those mentioned above, will be illuminated at a fixed dim value if they are turned on.

All of the light lamps can be tested at once by pressing the CAUTION LT TEST switch on the right-hand side of the caution panel. This will also test the NAV/GPS, Quiet Mode lamps (if installed). It will not test the pedal stop switch lamps. The pedal stop switch must be pressed (press to Test-PTT) to check its lamps.

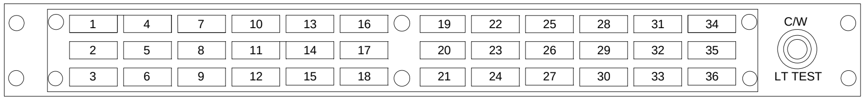


Table 1-1 Caution/Warning/Advisory Annunciators Matrix

Position	Annunciator	Color	Dimmable	Activation Polarity	Signal Condition
1	FLOAT TEST	Green	No	28VDC	FLOAT TEST SWITCH (24S7) PRESSED AND HELD WHILE FLOAT INFLATION SWITCH (24S9) IS ALSO PRESSED AND HELD
2	ENGINE FIRE	Red	Yes	28VDC	TEMPERATURE SENSORS DETECT A TEMPERATURE THAT INDICATES A PROBABLE FIRE.
3	ENGINE ANTI-ICE	Amber	Yes	GND	ENGINE ANTI-ICE SYSTEM SWITCH (4962-A35) ACTIVATED. (SWITCH B TO C CONTACT MADE ON INCREASING PRESSURE OF 5.5 +/- 0.5PSIG, BROKEN PRIOR TO 3.0 PSI DECREASING)
4	FLOAT ARM	Amber	Yes	28VDC	FLOAT ARM SWITCH (24S8) IN ARMED POSITION
5	FADEC CH FAIL	Amber	Yes	GND	ECU CHANNEL IN CONTROL ARINC WORDLABEL 270 BIT 19=1 (NG INDICATOR DISCRETE OUTPUT ACTIVATED)
6	START	White	Yes	GND	ECU CHANNEL IN CONTROL ARINC WORD LABEL 271 BIT 17=1 (NP/NR INDICATOR DISCRETE OUTPUT ACTIVATED)
7	BAGGAGE DOOR	Amber	Yes	GND	BAGGAGE DOOR SWITCH (4S6) CONTACTS CLOSED (BAGGAGE DOOR OPEN)
8	LITTER DOOR	Amber	Yes	GND	LITER DOOR SWITCH (24s6) CONTACTS CLOSED (LITTER DOOR OPEN)
9	HEATER OVERTEMP	Amber	Yes	28VDC	AIRCOMM BLEED AIR HEATER TEMP SENSOR ACTIVATED (SENSOR CLOSES AT 220 +/- 8°F (93 +/- 4.4°C)
10	L/FUEL BOOST	Amber	Yes	GND	MAIN FUEL CELL – LEFT FUEL PRESSURE SWITCH (1S22) ACTIVATED (SWITCH CONTACT MADE AT 1.5 +/- 0.5 PSI DECREASING, BROKEN AT 5 PSI MAX INCREASING)
11	L/FUEL XFR	Amber	Yes	GND	FWD. FUEL CELL – LEFT FUEL PRESSURE SWITCH (1S24) ACTIVATED (SWITCH CONTACT MADE AT 1.5 +/- 0.5 PSI DECREASING, BROKEN AT 5 PSI MAX INCREASING)
12	A/F FUEL FILTER	Amber	Yes	GND	AIRFRAME FUEL FILTER IMPENDING BYPASS INDICATOR SWITCH CLOSED (SWITCH CONTACT A TO B MADE AT 0.875 +/- 0.125 PSI)
13	R/FUEL BOOST	Amber	Yes	GND	MAIN FUEL CELL – RIGHT FUEL PRESSURE SWITCH (1S21) ACTIVATED (SWITCH CONTACT MADE AT 1.5 +/- 0.5 DECREASING, BROKEN AT 5 PSI MAX INCREASING)

Position	Annunciator	Color	Dimmable	Activation Polarity	Signal Condition
14	R/FUEL XFR	Amber	Yes	GND	FWD. FUEL CELL – RIGHT FUEL PRESSURE SWITCH (1S23) ACTIVATED (SWITCH CONTACT MADE AT 1.5 +/- 0.5 DECREASING, BROKEN AT 5 PSI MAX INCREASING)
15	FUEL VALVE	Amber	Yes	GND	FUEL SHUTOFF VALVE RELAY (1K6) DE-ENERGIZED (RELAY IS DE-ENERGIZED WHEN FUEL SHUTOFF VALVE (1B5) IS IN TRANSIT)
16	FADEC MAINT	White	Yes	GND	EITHER ECU CHANNEL ARINC WORD LABEL 350, 351, 352, 353, AND 354 BIT 14 THROUGH 29=1 ONLY WHEN NG <5% (NP/NR INDICATOR DISCRETE OUTPUT ACTIVATED)
17	START FAULT	White	Yes	GND	BOTH ECU CHANNEL ARINC WORD LABEL 351 BIT 21=1 (NP/NR INDICATOR DISCRETE OUTPUT ACTIVATED)
18	FUEL LOW	Amber	Yes	GND	LOW LEVEL DETECTOR (1S16) CONTACTS CLOSED (THIS WILL OCCUR AT APPROXIMATELY 100 +/- 10 LBS)
19	FADEC FAIL	Red	Yes	GND	BOTH ECU CHANNEL ARINC WORD LABEL 270 BIT 19=1 (NP/NR INDICATOR DISCRETE OUTPUT ACTIVATED) OR FADEC FAIL RELAYS (4962-K3 AND 4962-K4) BOTH ACTIVATED
20	FADEC DEGRADED	Amber	Yes	GND	EITHER ECU CHANNEL ARINC WORD LABEL 270 BIT 21=1 (NP/NR INDICATOR DISCRETE OUTPUT ACTIVATED)
21	ENGINE FUEL FILTER	Amber	Yes	GND	ENGINE FUEL FILTER IMPENDING BYPASS INDICATOR SWITCH CLOSED (SWITCH CONTACT A TO C MADE AT 0.875 +/- 0.125 PSI)
22	ENGINE CHIP	Amber	Yes	GND	GROUND PATH COMPLETED THROUGH ENGINE CHIP DETECTOR 4962-A2
23	XMSN CHIP	Amber	Yes	GND	GROUND PATH COMPLETED THROUGH TRANSMISSION CHIP DETECTOR (1E3, 1E5)
24	T/R CHIP	Amber	Yes	GND	GROUND PATH COMPLETED THROUGH TAIL ROTOR CHIP DETECTOR (1E4)
25	GEN FAIL	Amber	Yes	GND	GENERATOR RELAY (2K3) DE-ENERGIZED
26	XMSN OIL PRESS	Red	Yes	GND	TRANSMISSION OIL PRESSURE SWITCH (1S4) ACTIVATED (SWITCH CONTACT MADE AT 30 +/- 2 PSI DECREASING, BROKEN AT 38 PSI MAX INCREASING)
27	CHECK INSTR	Amber	Yes	GND	EXCEEDANCE HAS BEEN DETECTED BY ONE OF THE FOLLOWING INDICATORS: •TORQUE INDICATOR •MGT INDICATOR •NG INDICATOR •NP INDICATOR •ENGINE OIL PRESS & TEMP INDICATOR •FUEL PRESSURE AND AMMETER INDICATOR OR FAULT TO THE FUEL CONDITIONER UNIT HAVE BEEN DETECTED BY THE FUEL QUANTITY INDICATOR

Position	Annunciator	Color	Dimmable	Activation Polarity	Signal Condition
28	BATTERY RELAY	Amber	Yes	GND	BATTERY RELAY (2K1) IN ENERGIZED POSITION WITH BATTERY SWITCH (2S1) POSITIONED OFF
29	XMSN OIL TEMP	Red	Yes	GND	TRANSMISSION OIL TEMP SWITCH (1S3) ACTIVATED (SWITCH WILL ACTIVATE AT 230 +/- 10°F OR 110 +/- 5.6°C)
30	HYDRAULIC SYSTEM	Amber	Yes	GND	HYDRAULIC PRESSURE SWITCH (9S1) CONTACT A TO B MADE (A TO B CONTACT BROKEN AT 750 -0/+100 PSI INCREASING, MADE AT 650 +100/-0 DECREASING)
31	ENG OIL PRESS	Red	No	GND	ANNUNCIATOR COMES ON WHEN ENG OIL PRESS DROPS BELOW 42 PSI AND Ng > 80%
32	ENGINE OVSPD	Red	No	GND	ECU CHANNEL IN CONTROL ARINC 429 WORD LABEL 271 BIT 18=1 (NP/NR OR MGT INDICATOR DISCRETE OUTPUT ACTIVATED)
33	CYCLIC CENTERING	Amber	Yes	GND	THE ANNUNCIATOR COMES ON WHEN THE WEIGHT-ON-GEAR SWITCH (8S5) SENSES A WEIGHT-ON-GEAR CONDITION (HELICOPTER ON THE GROUND) AND THE CYCLIC STICK NOT CENTERED
34	ENGINE OUT	Red	No	GND	ECU CHANNEL INCONTROL ARINC 429 WORD LABEL 271 BIT 14=1 (NP/NR OR TORQUE INDICTAOR DISCRETE OUTPUT ACTIVATED)
35	PEDAL STOP	Amber	Yes	GND	INPUT POWER TO PRCU IS UNAVAILABLEOR INSUFFICIENT, INCORRECT SOLENOID POSITION, AIR DATA PRESSURE OUTPUT IS INVALID OR PRCU SELF TEST FAILURE
36	RPM	Red	No	GND	ROTOR RPM SENSOR SWITCH (4S2) ACTIVATED (E TO A CONTACT MADE AT 95 +/- 0.5% NR OR BELOW AND ABOVE 107 +/- 0.5 NROR HIGHER.

1.11 Power Plant

The HTS900-2-1D is a two-spool turboshaft engine (see Figure 1-7). A single-stage cooled gas producer turbine drives a two-stage centrifugal compressor and a single-stage power turbine drives the front-mounted reduction gearbox and the forward and aft power output shafts. Air is directed through a reverse-flow, effusion-cooled, annular combustor. The gas producer shaft is supported by a forward thrust bearing and an aft roller bearing. The power turbine is mounted on an aft thrust bearing and on two roller bearings at the front of the shaft. Main power drive is from the forward side of the gearbox. The output gear drives through an integral, one-way, sprag-type, overrunning clutch to the output shaft.

The engine is designed with a modular approach resulting in a total of three modules: the accessory/reduction gearbox, the gas producer, and the combustion/power turbine assembly.

The engine incorporates a dual-channel Full Authority Digital Engine Control (FADEC) system. The uninstalled engine has a takeoff power in the range of 820 shaft horsepower at standard day conditions. But due to the airframe limitation, the engine will provide 674 horsepower (560 ft-lbs) at takeoff power (100% instrument torque), and 630 horsepower (524 ft-lbs) at maximum continuous power (93.5% instrument torque).

1.12 Engine Controls – FADEC System

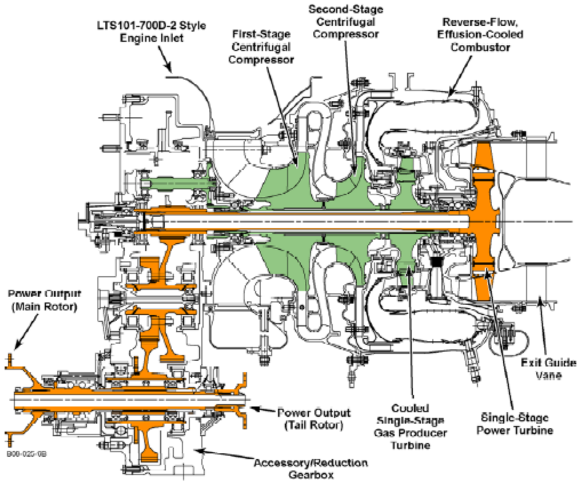
This paragraph addresses the operational design of the HTS900-2-1D FADEC system, its relationship with airframe and rotor systems, possible system faults, and troubleshooting. The information provided reflects operation with FADEC. Although Flight Manual Emergency Procedures are explained in this section, refer to the FMS-E407-789-1 Section 3, for actual flight operations.

1.12.A FADEC System

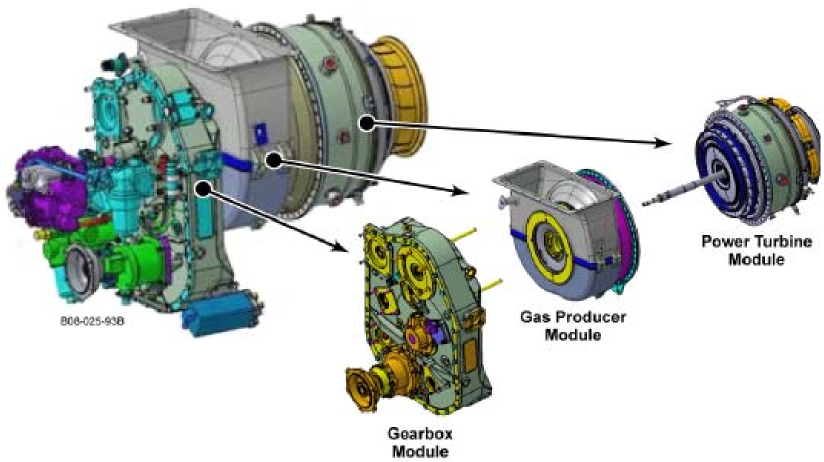
The FADEC system is designed to enhance flight safety and reduce pilot workload as well as provide other important benefit. In addition to the operational benefit of increased TBO, engine automatic start, and precise control of main rotor speed, the Eagle 407HP features redundant signal sensing, continuous monitoring, and self-diagnostics.

Although the possibility of a FADEC system failure is unlikely, pilots and maintenance personnel must have an operational understanding of the FADEC system, along with a sound knowledge of emergency and troubleshooting procedures. It is recommended that personnel involved with the Eagle 407HP familiarize themselves with the procedure for FADEC failure. Familiarization with this procedure will help to emphasize Flight Manual Emergency Procedures as the following FADEC information is read within Manufacturer's Data.

Respond to the horn and lights as described in Section 3, Emergency/Malfunction Procedures, in Flight Manual Supplement FMS-E407-789-1.



Major Engine Components



Major Engine Subassemblies

Figure 1-7 Honeywell HTS900-2-1D Engine

1.12.B FADEC System – Operation

The control system (Figure 1-8) is designed to operate in primary mode that provides capability of system full control and failed fixed mode when system encounters a dual-channel hard fault.

When operating in primary mode, the control system digitally executes all engine functions in response to airframe inputs and engine signals. Primary mode has full authority over fuel flow and automatic start scheduling.

The primary component of the FADEC system is an airframe-mounted dual-channel ECU. The ECU monitors numerous internal and external inputs to modulate fuel flow and therefore control engine speed, acceleration rate, temperature and other engine parameters. The ECU provides inputs to the HMU to modulate fuel flow based on the continuous monitoring of the following: Measured Gas Temperature (MGT), Gas Producer speed (NG), Power Turbine speed (NP), Main Rotor speed (NR), Engine Torque (Q), Collective Pitch (CP) and rate, Compressor Inlet Temperature (T1), Ambient Pressure (P0), Compressor Discharge Air Pressure (P3), and Power Lever Angle (PLA)/throttle position.

The engine mounted Hydromechanical assembly (HMA) consists of a fuel pump and a Fuel Metering Unit (FMU). Based on engine and cockpit signals, the ECU provides an electrical command to a dual stepper motor unit that sets the correct amount of fuel flow to the fuel ejectors, providing complete and automatic control of the engine during starting, steady-state, and transient operation throughout the engine operating envelope.

In case of a dual-channel hard-fault, the FMU stepper motor will be failed-fixed at its last commanded position, resulting in a continuous fixed fuel flow. Normal engine shutdown, ignition and starting are inhibited on a failed-fixed operation.

Engine shutdown during failed-fixed mode (emergency mode) can be achieved through the manual operation of fuel shutoff valve switch on the instrument panel to close the FMU and airframe shutoff valves.

1.12.C Power-Up Mode and Built-In Test

The FADEC system incorporates logic and circuitry to perform self-diagnostics. In general, sensors are checked for continuity, rate and proper range. Discrete inputs are checked for continuity and output drivers are monitored for current demand to sense failed actuators and open or shorted circuits. A FADEC power-up check exercises output drivers and actuators to ensure system functionality and readiness. The brief appearances of light indications and their respective horn observed immediately after application of power are normal and are part of the FADEC system's design initialization process. If any faults are detected during the self-test, the appropriate FADEC caution panel light will illuminate.

The helicopter 28 VDC bus supplies electrical power to the FADEC ECU until the engine achieves 45% Np. Above this speed, the FADEC ECU will select between the 28 VDC bus and the engine-driven permanent magnet alternator (PMA), as its primary power source. The higher voltage source will be selected. In the event of a primary power source failure, the alternate source will be selected.

1.12.D Start

For ground starting, the automatic start sequence is enabled by rolling the throttle PLA equal to or above the IDLE position and within 60 seconds, holding the momentary START switch for more than 0.5 seconds.

Although the start sequence is automatic, the pilot is responsible for monitoring the start and taking appropriate action required. Do not initiate a start if FADEC related caution panel lights are illuminated unless appropriate or successful corrective action has been carried out and no "current" faults are shown.

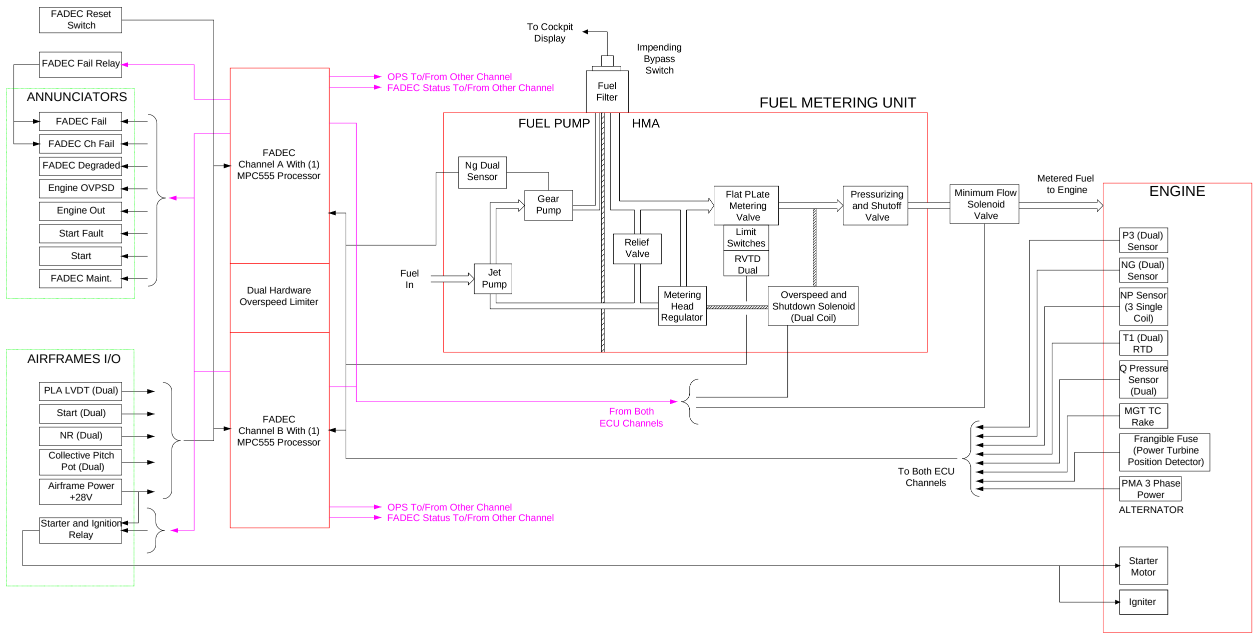


Figure 1-8 Control System Functional Diagram

NOTE

After the throttle is set to idle, the momentary contact start switch must be activated within 60 seconds to initiate the start and engage the latching feature. The latching feature of the start will engage when the FADEC ECU senses momentary activation (0.5 seconds) of the start switch. If a start is attempted following a delay of more than 60 seconds, the FADEC system will not allow the starter to latch following release of the start switch. Therefore, if a delay of more than 60 seconds has occurred, the system must be reset. To reset the system, the throttle must be repositioned to cutoff and then back to idle. In addition, if electrical power is interrupted prior to initiating the start, with the throttle in idle, the throttle must be repositioned to cutoff and then back to idle after power is restored to re-enable the latching feature. A normal automatic start sequence may then commence. If starting engine on external power, refer to paragraph 1.32.A, External Power.

To reduce residual MGT, a Dry Motoring Run may be performed in accordance with the Flight Manual Supplement FMS-E407-789-1 Section 2.

NOTE

ECU power –up testing will be terminated if the engine start sequence is initiated prior to the completion of BIT testing. If power-up testing is interrupted, FADEC related warnings, cautions, and advisories will not be displayed.

When the start switch is momentarily set to START, the FADEC/ECU senses the start signal causing it to energize the FADEC start relay of the channel in control. The FADEC start relay is then latched for the entire start sequence until the NG speed reaches 50%. During this time, the FADEC start relay allows the signal to the generator control unit/voltage regulator to inhibit generator output, flashes the shunt field for the duration of the start, and activates the starter relay. The starter relay activates the igniter relay. The igniter relay activates the engine igniter system. In addition, the FADEC/ECU transmits ARINC 429 data to the NP/NR engine indicator triggering the START advisory light on the Master Warning/Caution Panel.

Once NG speed reaches 10%, the FADEC system will introduce fuel, detect the lightoff, and smoothly accelerate the engine to idle while limiting MGT if necessary.

Upon reaching an engine NG speed of 50%, the FADEC/ECU unlatches the FADEC start relay and terminates the start sequence. The START advisory light should extinguish at this time.

The FADEC system also incorporates “Start Abort Logic” up to 50% NG during start. The control system will cut off fuel flow to the engine if any of the following conditions occur:

Start MGT exceeds 977°C (1790°F).

Supply voltage to FADEC/ECU drops below 12V. As a significant momentary voltage drop occurs at initiation of the start, ensuring a battery voltage of 24VDC or above prior to start, in conjunction with appropriate battery maintenance, will reduce the possibility of voltage dropping to 12VDC.

If NG speed does not reach 10% in 10 seconds.

If light-off does not occur within 35 seconds.

If idle speeds are not achieved in 60 seconds.

Engine acceleration from idle to 100% NR/NP, in primary mode, is achieved by smoothly increasing the throttle to the FLY detent position (PLA 70°). As the throttle is positioned from idle (30° to 40°) to the FLY detent position (PLA 70°), electrical signals are sent to the ECU from the PLA LVDT. These signals dictate the amount of authority the ECU has to control maximum fuel flow (NG limiting) based on throttle position, and in turn controls engine NG speed. Therefore, as the throttle is increased from idle to the FLY detent position, the fuel flow is electronically increased until 100% NR/NP is obtained. To avoid rapid engine acceleration, it is recommended that throttle application from idle to the FLY detent position be conducted in a smooth and gradual manner.

CAUTION

PERMANENT ENGINE DAMAGE MAY OCCUR IF ENGINE OIL TEMPERATURE IS NOT MAINTAINED AT OR ABOVE -10°F (-23°C) DURING COLD WEATHER STARTING.

1.12.E FADEC System Faults

Fault conditions are classified by the operating characteristics or capabilities of the engine control compared to the fully operational system in primary mode. Fault conditions that may occur are categorized below:

Maintenance Faults – These faults are transparent to the operator and would normally be indicated only after shutdown.

Degrade Faults – These faults are a result of an input to the control that may impact operation. The control system operates in a degraded condition but still in primary mode.

Hard Faults – Hard faults are a result of loss of inputs such that the ECU is incapable of controlling the engine. If a single ECU channel fail occurs, the healthy channel takes control of the system automatically. Should both channels be hard-faulted, the fail-fixed mode is enabled.

The FADEC/ECU provides ARINC 429 data to the cockpit instruments for engine monitoring and warning/caution annunciation. Both ECU channels are transmitting data simultaneously but the engine indicators determine the channel in control via bit 23 and 24 of ARINC data word label 270 and use valid data for indication.

There are eight lights in the caution/warning/advisory panel that are controlled by the FADEC and described as follows:

START- Indicates that the engine starter is engaged.

START FAULT – Indicates that the engine may not start.

FADEC MAINT – Indicates faults or failures in the FADEC related to redundancy or non-performance related features, or indicates that an engine incident has been logged by the ECU. This indication is provided after engine shutdown when NG is less than 5%.

FADEC DEGRADED – The active channel indicates a loss of function, resulting in degraded operation or performance, based on ARINC 429 data from the ECU channel in control.

FADEC CH FAIL –Indicates that an ECU channel has failed and that control was automatically swapped to the other channel, based on ARINC 429 data from the ECU channel in control.

FADEC FAIL – Indicates a dual-channel ECU hard fault and that the engine has failed to the last commanded fuel flow.

ENGINE OVSPD – Indicates that the engine control has detected an engine overspeed event.

ENGINE OUT – Indicates when engine NG drops below idle by 3% or uncommanded deceleration occurs for 0.5 seconds.

The FADEC ECU continuously monitors the FADEC system faults and makes appropriate accommodations to continue operation. Detected failures in the primary mode are accommodated in the following order of preference.

(1) Accommodate failure with no performance degradation via automatic switchover to the remote ECU channel.
(2) Accommodate failure with no performance degradation by using the signal from the remote ECU channel.
(3) Accommodate failure by disabling a noncritical function resulting in an acceptable level of performance degradation.
(4) Accommodate critical failure of a given channel that renders it incapable of controlling the engine, referred to as an ECU channel “hard fault”, by automatically transferring control to the other channel or, if both channels are hard faulted, by disabling the primary mode control and automatically reverting to the fail-fixed mode.

The FMS-E407-789-1 Section 3 provides the appropriate action required by the pilot for each light or light/horn condition.

1.12.F FADEC CH FAIL

Either of the FADEC/ECU channel should provide ARINC word label 270 bit 19 to the Gas Generator Indicator to activate ground discrete to the caution annunciator in the event of a single FADEC channel hard fault. Also, each ECU channel provides ground discrete parallel to the ARINC data signal for redundancy through individual FADEC fail relay.

1.12.G FADEC FAIL

Both FADEC/ECU channels should provide ARINC word label 270 bit 19 to the Dual Tachometer Indicator to activate ground discrete to the warning annunciator in the event of a dual FADEC channel hard fault. In addition, the FADEC FAIL relay will relax to the open position in the event of a dual FADEC channel fail, that would trigger the warning annunciator to activate in case of an ARINC data failure. Ground is also provided to the FADEC fail horn which is not mutable.

1.12.H ENGINE OVSPD

An engine overspeed condition exists when the FADEC/ECU detects overspeed on the gas producer turbine (NG) or power turbine (NP). The

ENGINE OVSPD warning annunciator comes on when the FADEC/ECU channel in control transmits ARINC 429 data bus word label 271 bit 18 set to switch discrete ground on the Dual Tachometer Indicator (NP/NR) or the Measured Gas Temperature Indicator (MGT).

1.12.I ENGINE OUT

An engine out condition exists when NG drops below idle by 3% or an uncommanded deceleration occurs for 0.5 seconds. The ENGINE OUT warning annunciator comes on when the FADEC/ECU channel in control transmits ARINC 429 data bus word label 271 bit 14 set to switch discrete ground on the Dual Tachometer Indicator (NP/NR) or the Torque indicator (Q). It also provides ground to the engine out horn. The engine out horn can be muted by the horn mute button on the instrument panel.

1.12.J FADEC MAINT

The FADEC MAINT light is triggered by any Confirmed Fault Word (FADEC ARINC label 350 through 354 Bits 14 through 29) set equal to “1”. During this time, the control system operates normally with no degradation. The FADEC MAINT light indicates maintenance is required and is displayed only on the ground after engine shutdown with NG less than 5%.

1.12.K FADEC DEGRADED

The following table describes engine control faults that result in a degraded engine control state. In this state, the type or degree of engine control functionality lost depends on the type of the fault that initiated the FADEC DEGRADED mode. The effects of some of these faults may not be immediately apparent, while other faults may have a noticeable effect on engine control.

Table 1-2 FADEC Degraded Faults

Fault	Effect
Nr fault	Fault results in degradation of autorotation and loss of load logic.
MGT fault	Fault results in loss of MGT limiting. Temperature limiting during starting and operation is disabled.
Ambient temperature fault	Fault results in engine accelerations and decelerations degraded to ISA conditions based on ambient pressure. Only cold-day light-off fuel flows will be lower than normal. Cold day starts may or may not be successful.
Ambient pressure fault	Fault result in engine accelerations and decelerations degraded based on default value of P0, 12.23 psia. This is equivalent to 5,000 feet pressure altitude. Starts above 5,000 feet pressure altitude may be hotter than normal and may or may not be successful.
Compressor discharge pressure fault	Fault results in the loss of a level of minimum and maximum acceleration and deceleration limiting based on fuel ratio units.
Engine-torque fault	Fault results in loss of engine torque limiting.
Starter relay fault	The engine cannot be started.
PLA fault	The engine continues to operate only in its current operating condition (for example, OFF, IDLE, or NP governing)
Extended time in freeze-state fault	May not be observable. ECU has been frozen too long confirming a fault. A channel swap will occur.
10-bit QADC A fault	May not be observable. This is an analog-to-digital conversion fault. May precipitate other listed faults.
10-bit QADC A fault	May not be observable. This is an analog-to-digital conversion fault. May precipitate other listed faults.
Collective pitch fault	Loss of collective pitch anticipation. Increase rotor droop may be experienced.
Cross-channel data link fault	If present when an input signal is lost to the channel in control, the channel in control will change. Otherwise, the fault is not observable.
Start switch fault	The start switch is faulted on both channels. The engine cannot be started.
Fuel flow metering valve cross-channel comparison fault	The calculated fuel flow does not agree across ECU channels. Engine may operate on minimum or maximum limiting schedules more often. A noticeable change in engine operation is not expected.
Overspeed protection system fault	The OPS has a continuous monitor fault.
Overspeed protection system frequency power-up test fault	The OPS failed the frequency power-up test when power was initially applied to the ECU. An ECU channel attempted to drive the MFV and shutoff valve (SOV), and an electrical fault resulted on one or both component.

Fault	Effect
Overspeed protection system inhibit feedback fault	The OPS inhibit feedback is faulted.
Minflow solenoid IBIT fault	The minflow solenoid failed a current BIT check. An ECU channel cannot control the minflow solenoid.
Shutoff solenoid IBIT fault	The SOV failed a current BIT check. An ECU channel cannot control the shutoff solenoid.
Shutoff solenoid low-side switch VBIT fault	The shutoff solenoid low-side switch is faulted. An ECU channel cannot control the shutoff solenoid.
Shutoff solenoid high-side switch VBIT fault	The shutoff solenoid high-side switch is faulted. An ECU channel cannot control the shutoff solenoid.
Minflow solenoid low-side switch VBIT fault	The minflow solenoid low-side switch is faulted. An ECU channel cannot control the minflow solenoid.
Minflow solenoid high-side switch VBIT	The minflow solenoid high-side switch is faulted. An ECU channel cannot control the minflow solenoid.
OPS minflow solenoid fault during shutdown test	The minflow solenoid did not pass the last shutdown test.
OPS shutoff solenoid fault during shutdown test	The shutoff solenoid did not pass the last shutdown test.

1.12.L FADEC RESET Switch

The FADEC RESET switch is installed on the instrument panel and available to the pilot to reset FADEC faults. Toggling the reset switch instructs the ECU to attempt to clear/reset any faults that might be present in the ECU. Stable flight conditions are recommended when attempting to reset ECU faults in flight.

1.12.M Engine Shutdown

The control system allows the engine to shutdown manually by either throttle command or by using the emergency fuel valve switch. The control system also has the ability to automatically shutdown the engine as a result of a start-abort condition. The engine incorporates two ways to shutoff fuel completely to the engine combustion system: the FMU shutoff solenoid and the FMU metering valve.

Normal manual shutdown is initiated from the cockpit by selection of the throttle to the OFF position. Following stable engine operation for more than 3 seconds at ground idle, a shutdown test is performed. The minflow valve and fuel shutoff solenoid in the Fuel Metering Unit are enabled and functionally monitored, followed by the metering valve commanded to 0 lb/hr.

Motoring after engine shutdown is required to reduce turbine sump temperature. Failure to motor engine after shutdown may require additional maintenance action. Refer to Honeywell LMM for HTS900-2-1D.

1.12.N Emergency Shutdown

The control system features a pilot-activated emergency fuel valve switch that is directly connected to the shutoff solenoid on the FMU and to the ECU (for feedback confirmation of a pilot-commanded emergency shutdown and for fault detection of the emergency fuel valve switch). No engine starts can be initiated by the control system while the emergency fuel valve switch is activated.

1.12.O Checking FADEC Fault Codes

Faults can be displayed immediately via FADEC FAIL, FADEC CH FAIL, START FAULT and FADEC DEGRADED or by a combination of these lights. Maintenance advisory faults will be displayed on shutdown via the FADEC MAINT light. In addition, faults are also used to identify exceedances and incidents recorded by the ECU.

Regardless if the fault light(s) were displayed in–flight or at shutdown, maintenance action is required prior to further flight. For maintenance troubleshooting, please refer to ICA-E407-789 and Honeywell LMM for HTS900-2-1D.

The preferred method of determining FADEC faults is with the Ground Support Equipment (laptop computer). The GSE is capable of providing information on the ECU's live mode (current faults) and stored mode faults. The ECU's fault snapshot data can be uploaded and saved. This data is ordered and displayed in increasing engine run time to assist in troubleshooting. Additionally, event information regarding the incident recorded by the ECU can be uploaded to the laptop for viewing or saving. All events are sorted based on the engine run time. Engine history data can be uploaded and downloaded to the ECU if the control is transferred to another engine. Engine exceedances monitored by the control system can also be uploaded for viewing or saving.

Fault codes will be displayed on the Fault History page of the GSE. After determining the fault code, refer to Honeywell LMM for HTS900-2-1D for corresponding maintenance action required. If a system fault has been observed that does not result in a fault code, record the symptom(s) and proceed with engine troubleshooting based upon the operational problem.

1.12.P Clearing FADEC Fault Codes

Faults are not to be erased unless appropriate maintenance actions have been carried out in accordance with ICA-E407-789 and Honeywell LMM for HTS900-2-1D. Do not attempt to clear any fault or exceedances while the engine is operating.

If maintenance actions have been conducted due to a recorded exceedance or to correct a current or last engine run fault, it must be ensured that no FADEC system lights are illuminated when the throttle is positioned to idle for the next start attempt. If a FADEC related light is illuminated with the throttle positioned to IDLE, a current fault exists and further maintenance action is required.

After completing the maintenance actions, reset the FADEC fault by pressing FADEC RESET switch for a minimum of 10 seconds and wait 15 seconds minimum.

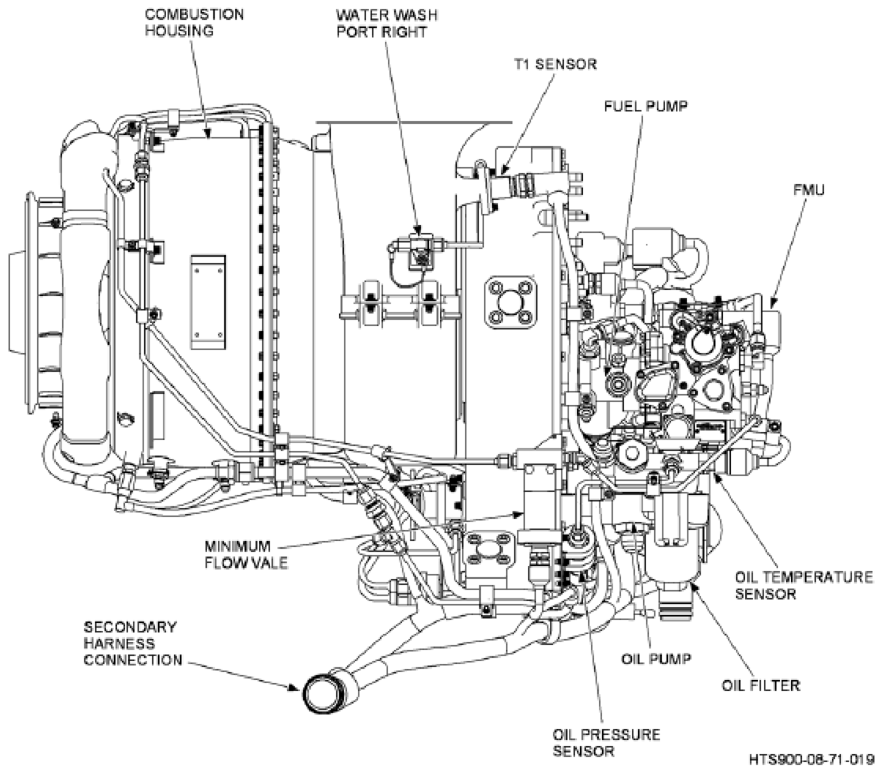


Figure 1-9 Power Plant Components (Sheet 1 of 4)

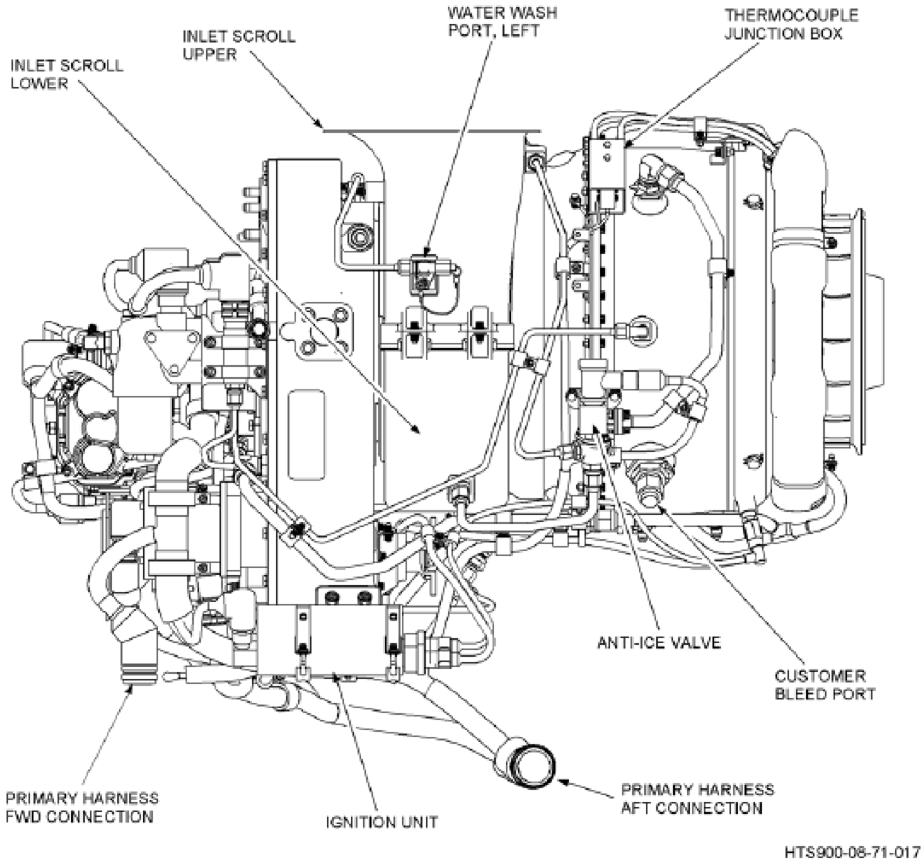


Figure 1-9 Power Plant Components (Sheet 2 of 4)

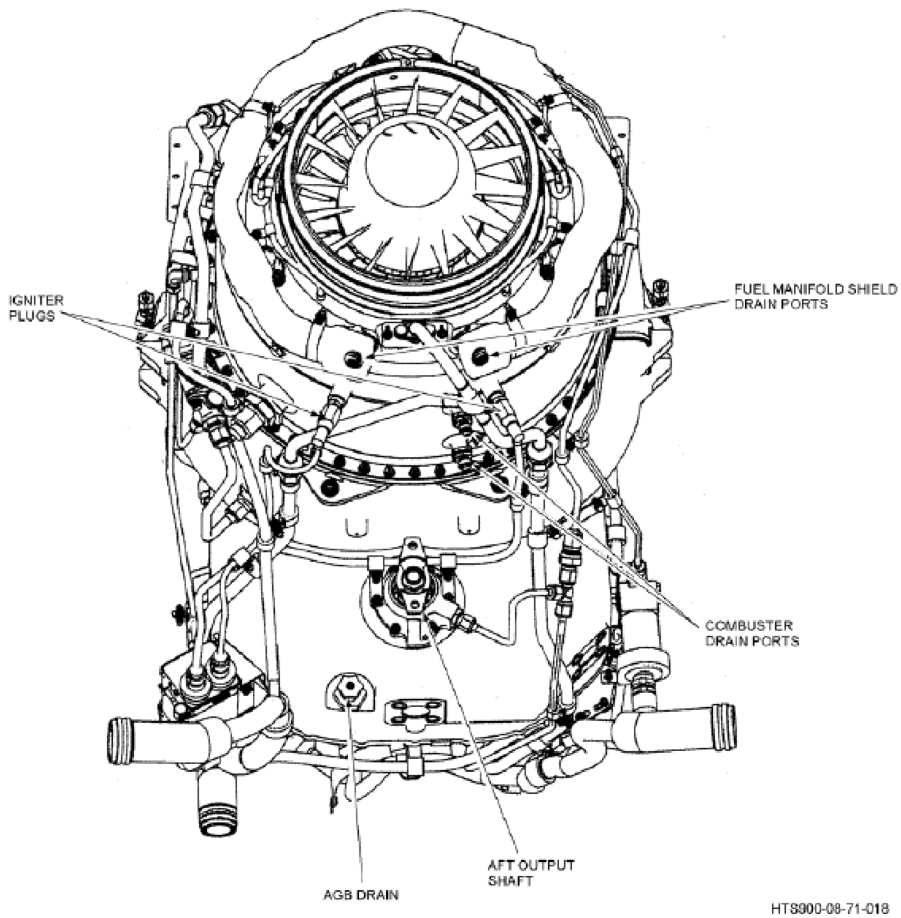


Figure 1-9 Power Plant Components (Sheet 3 of 4)

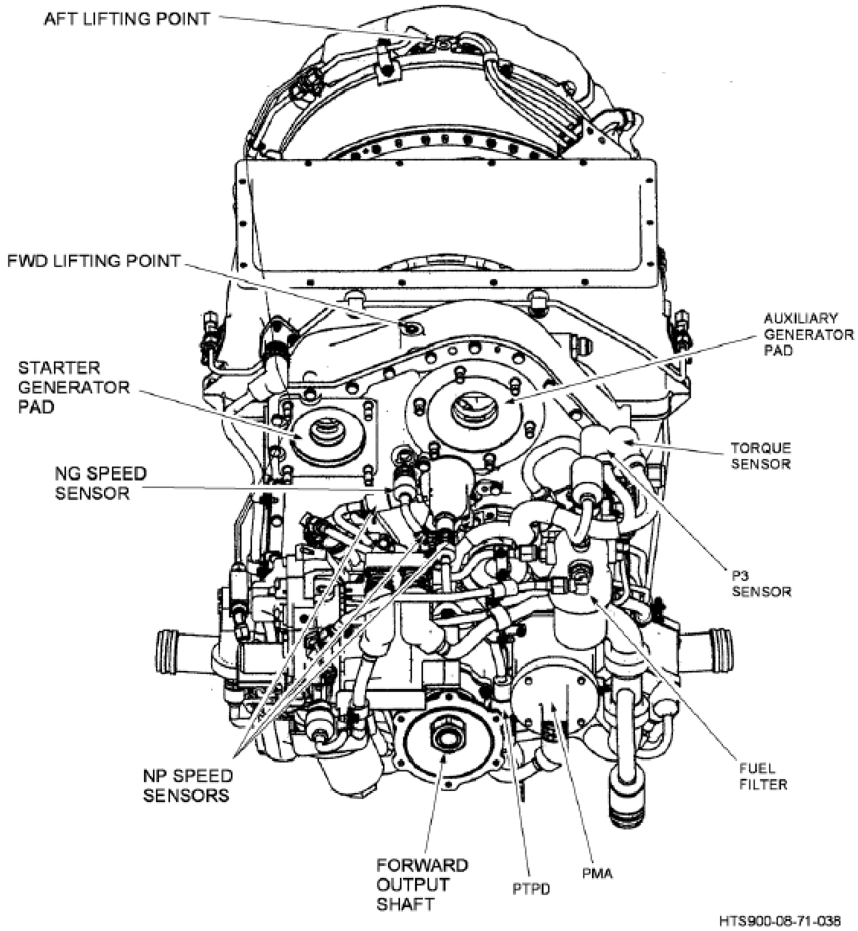


Figure 1-9 Power Plant Components (Sheet 4 of 4)

1.12.Q Exceedances Recorded by ECU

Although engine exceedance parameters are monitored and recorded by individual gauges, the FADEC/ECU also records predetermined values of exceedances. The FADEC/ECU values are for reference only and can only be used for maintenance troubleshooting. Exceedances recorded by the gauges are always to be used for determining the required maintenance action. Clearing an exceedance from the FADEC/ECU using the GSE is only to be performed if operating time above a limit has triggered a maintenance action and the maintenance action has been completed.

1.13 Deleted

1.14 Powerplant Ignition System

The ignition system consists of a dual-channel, medium tension exciter, two output leads and two igniter plugs. Both ignition channels are powered by 28 VDC when the starter is energized.

The ignition system transfers energy to the combustible fuel mixture. Energy is provided in the form of high temperature – high amperage arcs at the spark igniter gaps. These arcs ignite the fuel/air mixture. The ignition exciter is only required during the starting cycle since the combustion process is continues. Once ignition takes place, the flame in the combustion liner acts as ignition agent for the fuel/air mixture.

1.15 Powerplant Temperature Measuring System

The temperature measuring system consists of four three-probe Chromel-Alumel thermocouple assemblies and an averaging junction box. Closed and ungrounded sheathed thermocouples are used in the gas flow path. A transition is then made to a high-temperature flexible lead, which extends forward on the engine to the junction box. The indicated gas temperature includes an empirically derived bias to account for differences between the thermocouple measurement and the thermodynamic gas producer turbine temperature.

1.16 Powerplant Compressor Bleed Air System

The compressor bleed airflow is available for aircraft use at all engine operating levels and during all flight conditions, except during starting.

The engine is provided with a non-removable orifice to limit bleed flow to a level that does not produce an unsafe condition for the engine. The permanently installed orifice, combined with the interface fitting, limits the bleed discharge to 5.6 % at takeoff conditions, preventing engine damage in the event of a failure within the bleed system.

During customer bleed air extraction, pilot monitoring action may be required to maintain engine NG and MGT within engine operating limits.

1.17 Engine Oil System

The engine oil system incorporates a dry sump type lubrication system. The engine oil system includes an externally mounted oil tank, a temperature bulb, a manually operated drain valve, and oil cooler. The oil cooler is installed on top of the fuselage, behind the aft firewall. The lubrication oil is supplied to and from the engine through rigid and flexible tubes. Refer to Figure 1-10.

The lubrication pump draws oil from the tank and discharges it into the engine oil filter. The oil filter is equipped with a bypass valve if the filter becomes clogged. A mechanical indicator alerts the operator to impending filter bypass.

Before the filtered oil flows into the engine, it passes through the low-pressure shutoff valve. The spring load on this valve is sized to close at low speeds, which prevents oil from flowing into the engine during freewheeling.

The oil system is regulated through an externally adjustable pressure-regulating valve. Oil is removed from the turbine sump and gearbox sump with a scavenge pump. The turbine sump vent line removes air from the sump and delivers it to the gearbox, where the air is vented overboard through a dynamic air/oil separator.

Description	Interface Point
Impending oil bypass indicator	SL1
Accessory gearbox drain plug	SL2
Oil pressure adjustment valve	SL3

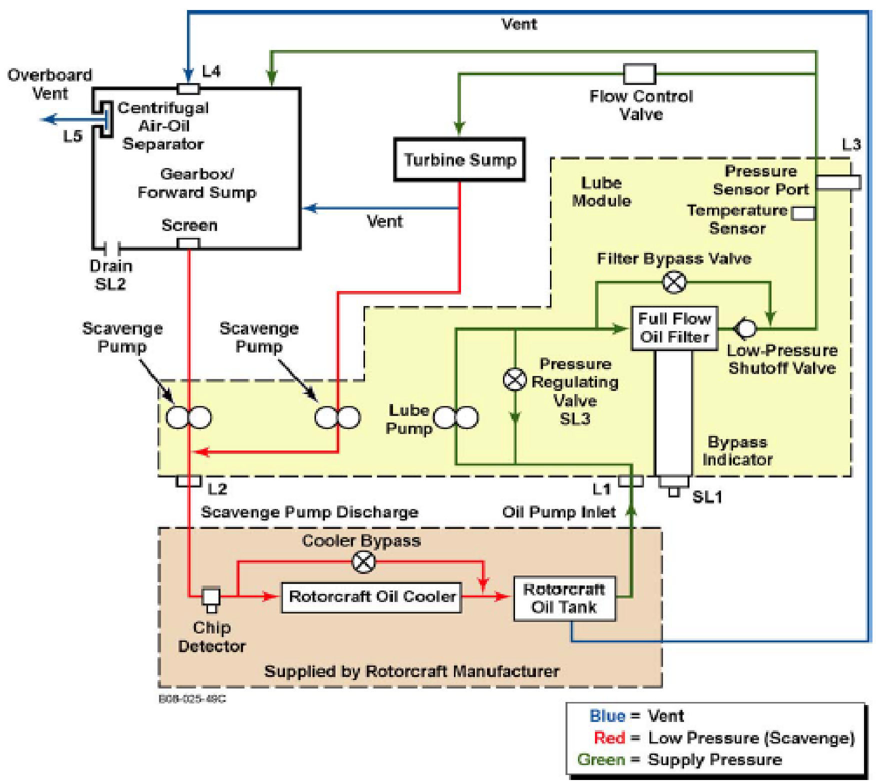


Figure 1-10 Engine Oil System

From the scavenge pump, oil will pass through the chip detector and into the oil cooler.

The system is designed to furnish adequate lubrication and cooling flow to the engine bearings, gears, seal rotors, splines (including the starter pad spline), and power output shafts.

The approximate capacity of the engine oil tank is 1.5 US gallons, and the oil level is checked by means of a sight gauge mounted on the left hand side of the tank. Viewing access to the sight gauge is provided by a cutout in the cowling. The oil cooler is mounted on top of the duct on the oil cooler blower.

1.18 Deleted

1.19 Deleted

1.20 Engine Anti-Ice Switch

The ENG ANTI-ICE switch, located on the overhead console, controls the engine anti-ice bleed air solenoid valve. The engine anti-ice system will be activated when the ENG ANTI ICE switch is positioned to ENG ANTI ICE. This de-energizes the engine anti-ice bleed air solenoid valve allowing hot diffuser scroll air to flow from the engine anti-icing valve to the engine compressor front support guide vanes and prevent the formation of ice. In the event of a total electrical system failure, the anti-ice system will fail safe to ON and provide continuous anti-icing.

When the ENG ANTI ICE switch is positioned to OFF, bus voltage is provided to the engine anti-ice bleed air solenoid valve from the engine anti-ice circuit breaker. This energizes the engine anti-ice bleed air solenoid valve ad prevents the flow of hot air from the engine anti-ice valve to the engine compressor front support guide vanes.

Since the Eagle 407HP incorporates an ENGINE ANTI ICE caution panel annunciator, an engine anti-ice pressure switch is used to control activation of the annunciator. Engine anti-ice pressure switch activation occurs on increasing pressure at 5.5 +/- PSI (37.9 ±3.4 kPa) which allows the ENGINE ANTI ICE annunciator to illuminate. Engine anti-ice pressure switch deactivation occurs on decreasing pressure prior to 3.0 PSI (20.68 kPa), which turns OFF the ENGINE ANTI ICE annunciator.

1.21 Engine Indicators

Propulsion instruments provide an indication of the performance of the systems related to the power plant. Engine indicators include a Torque, NG, MGT, NR/NP, and Engine Oil Temperature/Pressure gauge.

1.21.A Torque Gauge

The engine torque system is composed of an engine torque indicator and a single dual element engine torque transducer. The torque sensor is mounted on the front of the gearbox and has redundant strain gauges (one output per channel) that measure oil pressure proportional to engine torque.

The torque indicator is powered by 28VDC through its individual circuit breaker located on the overhead circuit breaker and switch panel.

1.21.A.1 Torque Gauge Range Marking

The 100% maximum on the torque instrument represent 560 ft-lbs torque (674 Shaft horsepower at 100% Np). This output is divided on the helicopter between the tail rotor drive system and the main rotor drive system. This division of horsepower is done automatically in flight, depending on the power requirements of the different drivetrain systems.

The 93.5% on the torque gauge represents 524 ft-lbs torque (630 shaft horsepower at 100% Np). This is the maximum limit allowed on a continuous basis. The 5-minute limit between 93.5% (524 ft-lbs) and 100% (560 ft-lbs) represents a 5-minute limit in this range of power. The torque recorded by the indicator is different from the torque recorded by the ECU.

In the event of a recorded torque exceedance, refer to both the ICA-E407-789 and Honeywell LMM for HTS900-2-1D.

NOTE

100% indicated torque is different from 100% engine torque recorded by the ECU. Due to the physical restriction on the main transmission, the indication of 100% torque equates to it's continuous operational capability. Thus, the ARINC 429 signal transmitted by the FADEC/ECU for torque indication has been conditioned to get the gauge to read 100% at 560 ft-lbs. To determine the engine (FADEC/ECU) torque, use the factor (Qx 0.8903) or (Q/1.123). When evaluating an engine overtorque situation, the mentioned information will be valuable in relating the torque indication system to the relevant engine torque values.

1.21.A.2 Engine Torque Exceedance

The torque indicator microprocessor is pre-programmed with specific torque values, which causes the digital display to flash.

The peak torque and duration of transient torque events during operation are recorded in the memory of the indicator. The duration of each event is captured by timers which are started when the torque exceeds the torque limits given in Table 1-3.

The indicator counts over-torque events by maintaining counters in its non-volatile memory. The Operation Exceedance Count (OEC) counts the number of over-torque events during the normal operation.

There is a torque limitation of 5 minutes between 93.5% and 100%. To give the pilot notice that he is approaching the end of the 5 minute limit, the indicator's digital display begins to flash (once per second) when 30 seconds remain in the 5 minute time period. The display stops flashing when indicated torque returns below the alert limit as long as an over-torque event did not occur.

There is a redline at 100% representing the maximum torque allowed. When the pilot exceeds 100%, the indicator's digital display flashes immediately twice per second. The display stops flashing when indicated torque returns below the alert limit as long as an over-torque event did not occur.

There is a transmission inspection limit at 110%. When the pilot exceeds 100%, the indicator immediately flashes (twice per second). The CHECK INSTR annunciator activates to indicate that an exceedance has occurred and has been recorded in the indicator's Non-Volatile Memory. The CHECK INSTR annunciator is activated until the display sequence is completed.

Table 1-3 Engine Torque Exceedance Monitoring

Torque%	Indication	Exceedances
93.5 to 100%	Digital display begins flashing once per second after 4.5 minutes.	Not recorded.
100.0 to 110%	Digital display immediately flashes twice per seconds.	Not recorded.
Above 110.0%	Digital display immediately flashes twice per second. CHECK INSTR annunciator activated.	Recorded immediately.

1.21.B Gas Producer Gauge (NG)

The gas producer (NG) gauge displays engine gas producer speed in percent of rated RPM (Table 1-4). It is powered by 28VDC through its individual circuit breaker located on the overhead circuit breaker and switch panel.

The engine includes two independent NG sensors that are both dual-coil sensors that measure the rotational speed of the gas producer shaft. The primary NG signals for each ECU channel are obtained from a pickup located on the front of the engine gearbox. The redundant speed signals are obtained from the second monopole mounted on the FMU. With data sharing via the cross-channel data link, each ECU channel accesses both NG signal.

The Ng indicator monitors last flight gas generator cycles and displays a value when NG is below 5%. The recorded last flight cycle count is displayed in a format of C#.## or C##.#. When the generator speed is above 55%, the recorded cycle data is updated.

The indicator also monitors the data bus for words with label 271 for the FADEC channel fail caution (bit 19). If either of the channels indicates FADEC fail, the FADEC CH FAIL caution annunciator output is activated.

Table 1-4 Gas Producer Gauge (NG) Exceedance Monitoring

NG%	Indication	Exceedance
101.1-103.6%	Digital display begins flashing once per second after 4.5 minutes	Not recorded.
Above 103.6%	Digital display immediately flashes twice per second. CHECK INSTR annunciator activated.	Recorded immediately.

1.21.C Dual Tach Gauge

The Dual Tachometer has two pointers: one that displays Main Rotor RPM on the outer scale and one that displays engine NP on the inner scale. All scales are in percent of rated RPM. The indicator is powered by NP and NR circuit breakers in parallel with each other. An isolation diode is installed between the two power supplies.

Three gearbox-mounted NP sensors are used to measure the rotational speed of the power turbine. Each ECU channel has a dedicated NP signal, and one is shared between the channels.

An NR speed pickup mounted on the transmission lower case provides three separate identical signal outputs of rotor RPM. Two signals are sent to each FADEC/ECU channel and one signal is sent to the Main Rotor RPM sensor switch.

During LCD test, both needles turn to indicate 100% only when the NG is lower than 5%.

The peak NP speed and duration are recorded in memory. The duration of each event is captured by timers which are started when NP exceeds the limits of 105.0% or 115.0%.

The indicator has eight outputs which provide switch grounds. Annunciator outputs activate when the specific condition is detected on the ARINC 429 data bus relevant to FADEC/ECU control system. See Table 1-5.

Table 1-5 Dual Tachometer Discrete Output

ARINC Label	Bit	Channel	Indications Activated		Conditions
270	21	Either	FADEC DEGRADED	Caution	FADEC degraded faults detected per section 1.12.K
270	19	Both	FADEC FAIL	Warning	Both ECU channels hard faulted
271	18	Active	ENGINE OVSPD	Warning	Engine control detected NG or NP overspeed.
271	17	Active	START	Advisory	Engine start.
271	14	Active	ENGINE OUT	Warning	Engine drops below idle.
351	21	Both	START FAULT	Advisory	Engine may not start.
351-354	14-29	Either	FADEC MAINT	Caution	FADEC maintenance fault detected as per Chapter 76-00-00. Will annunciate with NG below 5%.
			CHECK INSTR	Caution	NP exceedance detected.

Table 1-6 Dual Tachometer Gauge (NP) Exceedance Monitoring

NP%	Indication	Exceedance
Above 105.0% or Above 115%	CHECK INSTR annunciator activated.	Recorded immediately.

1.21.D Measured Gas Temperature

The measured gas temperature (MGT) gauge displays engine gas temperature of air between the gas producer turbine and power turbine in degrees Celsius.

The peak temperature and duration of transient events during operation are recorded in the memory of the indicator. The duration of each event is captured by timers which are started when the temperature exceeds the limits given in Table 1-7. The indicator monitors last flight power turbine cycles and displays a value when NG is below 5%. The recorded last flight cycle count is displayed in a format of C#.## or C##.#. When the generator speed is above 55%, the recorded cycle data is updated.

The indicator monitors the data bus for words with label 271 for overspeed event warning (bit 18). If the active channel indicates overspeed, the ENGINE OVSPD annunciator is activated.

Table 1-7 MGT Exceedance Monitoring

MGT°C	Indication	Exceedance
900°C-958°C	Digital display begins flashing once per second after 4.5 minutes	Not recorded.
Above 958°C or Above 977°C	Digital display immediately flashes twice per second. CHECK INSTR annunciator activated.	Recorded immediately.

1.21.E Engine Oil Temperature/Pressure Gauge

The engine oil temperature and pressure gauge is a dual instrument that simultaneously displays oil temperature in degrees Celsius on the right side display and oil pressure in PSI on the left hand side display. The engine oil temperature signal is provided by a thermo bulb installed on the engine. The engine oil pressure is provided by a transducer mounted on the forward engine firewall. When engine oil pressure drops below 42 psi and Ng > 80%, the ENG OIL PRESS warning light is turned on.

1.22 Engine Out Warning Light and Horn

The ENGINE OUT light and (pulsing) warning horn circuit will activate when the FADEC detects an engine flameout and ARINC 429 data bus Word 271 bit 14 is set to 1. The engine out (pulsing) warning horn can be muted by pressing the HORN MUTE switch.

1.23 Engine Overspeed Warning Light

The ENGINE OVSPD light will illuminate if the FADEC detects an NG overspeed of 110.4% RPM, or when NP overspeed is detected. FADEC/ECU NP overspeed has two conditions to meet in order to declare an overspeed event (1) when NP exceeds 105.4% RPM and (2) where % RPM is higher than the set speed for specific rate of change. This is defined by a linear relationship between the NP signal and the rate of change.

Once the overspeed event is no longer present, the annunciator will extinguish and the incident will be recorded in the ECU.

If the ENGINE OVSPD light is activated during engine operation due to an exceedance, it will be recorded by the ECU and the pilot will be provided with a maintenance advisory FADEC MAINT or by the caution light FADEC DEGRADED. Maintenance action is required prior to further flight. Peak values of exceedances are recorded on specific gauges. Any faults related to the exceedances are recorded in the ECU and can be retrieved on the engine history page of the Ground Support Interface.

1.24 Engine Chip Detector Light

The engine chip light will illuminate if metallic particles bridge the gap of the engine chip detector element. The magnetic chip detector is mounted on the engine chip detector housing downstream of the scavenge pump discharge port ahead of the oil cooler.

1.25 Fuel System Description

The fuel system (Figure 1-11) consists of two crash resistant, bladder type fuel cells. The forward fuel cell is located underneath and between the aft facing passenger seats. The aft fuel cell is located underneath and behind the aft passenger seats.

Both fuel cells are serviced through the filler port located on the right side of the helicopter. Approximately 28.4 US gallons (193.1lb) of fuel will accumulate in the aft main fuel cell, prior to the forward cell being filled through the gravity feed stand pipe. The gravity feed stand pipe connects the aft fuel cell to the forward tank. As the aft fuel cell fills beyond the level of the top of the stand pipe, the forward tank will be completely filled (forward tank fuel capacity 37.6 US gallons (256.0 lb). The aft tank will then be filled to the level of the filler port (usable fuel capacity 127.8 US gallons (869.0 lb).

If the auxiliary tank is installed, it will be filled at the same time as the aft tank through two openings located on the lower aft wall of the main fuel cell, which are connected to the bottom of the auxiliary tank (refer to BHT-407-FMS-6 for information on this kit).

Fuel from the forward tank is transferred to the main tank by two transfer pumps mounted on a sump plate assembly located on the bottom of the forward fuel cell. Fuel from the main fuel cell will be supplied to the engine through two boost pumps located at the base of the main fuel cell on a sump plate. The fuel from the two boost pumps joins into a common fuel line that passes through a fuel shutoff valve, then through an airframe mounted fuel filter before reaching the engine driven pump on the HMU. A solenoid sump drain valve is installed on the sump plate assemblies of both the forward and aft fuel tanks. The drain valves are activated by two switches located on the right hand side of the lower fuselage. These switches are deactivated when the fuel valve switch is ON to prevent inadvertent activation during flight.

1.25.A Fuel System Operation

With power applied to the helicopter and the Fuel XFR/Boost Right and Fuel XFR/Boost Left circuit breaker switches ON, the two transfer pumps and the two boost pumps are operating. Refer to Figure 1-12 for the fuel transfer system schematic.

Each transfer pump sends fuel through a one-way check valve located at the outlet of each pump and a common tee fitting through a line which transfers the fuel into the aft tank. Each check valve will assure that if either of the transfer pumps becomes inoperative fuel will be pumped to the aft tank and not through the inoperative pump back into the fwd fuel tank.

A modified tee fitting incorporating an orifice allows for a specific amount of fuel to bleed back into the forward tank to reduce the transfer rate.

Each boost pump sends fuel through a one-way check valve located at the outlet of each pump and a common tee fitting. It then flows through a common line to a fitting located at the top right side of the main fuel cell. The line leaves the fuel cell and passes through the fuel shut off valve. A pressure transducer (located between the main fuel cell fitting and the fuel shut off valve) supplies the electrical signal to the instrument panel mounted fuel pressure gauge.

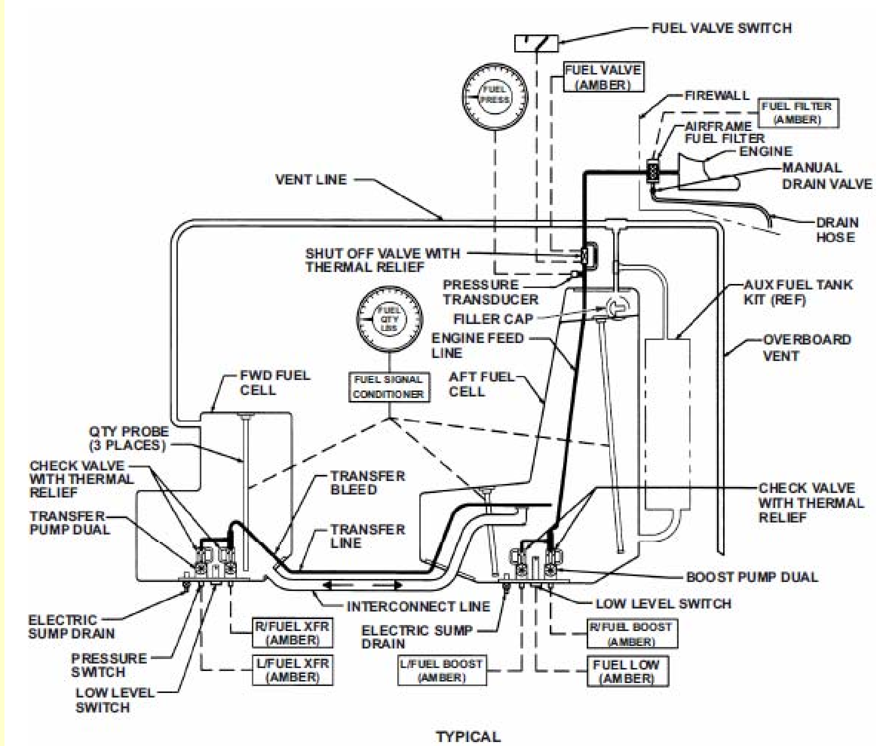


Figure 1-11 Fuel System

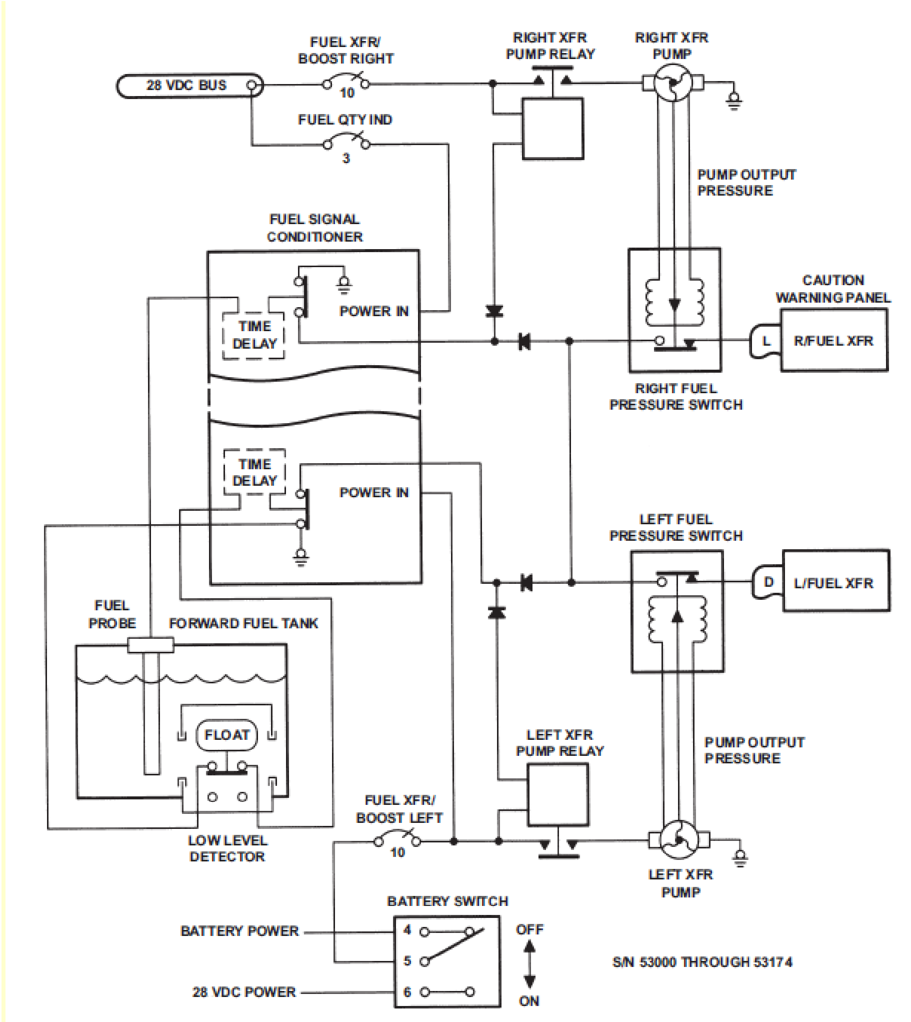


Figure 1-12 Fuel Transfer System Schematic (Sheet 1 of 2)

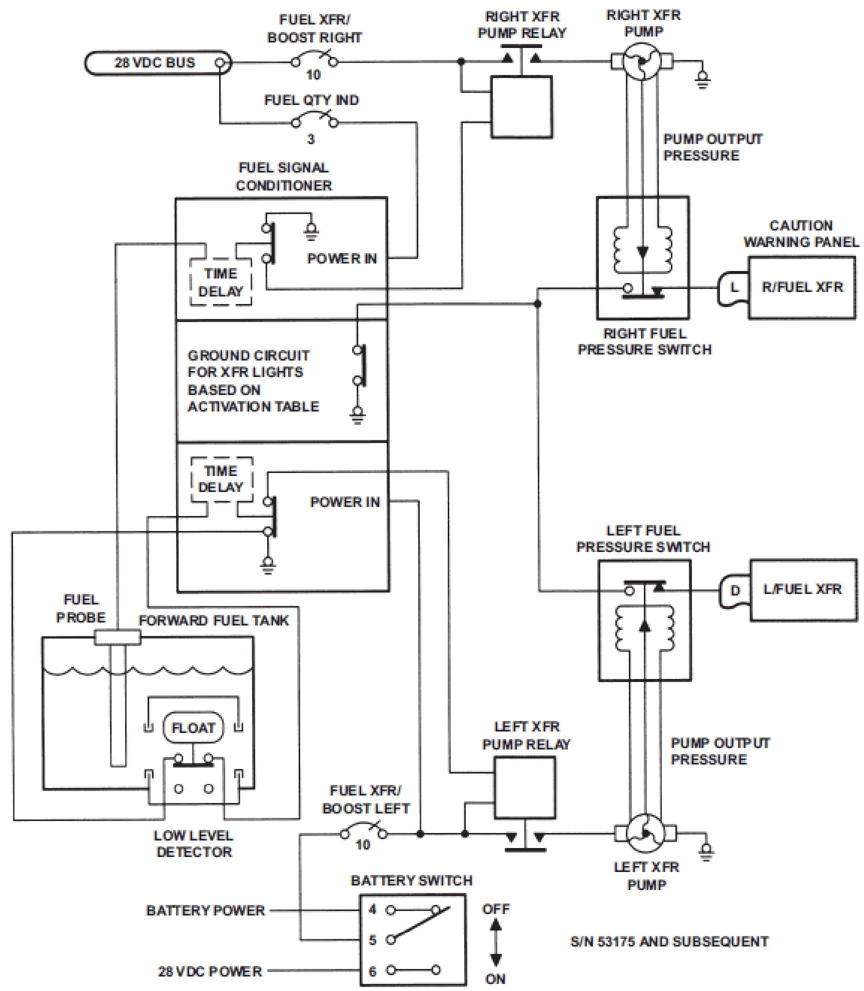


Figure 1-12 Fuel Transfer System Schematic (Sheet 2 of 2)

During normal operation, fuel will first be used from the aft fuel cell until its level is equivalent to the top of the forward tank. At this time, the aft cell and forward cell will be used equally down to the level of the top of the gravity feed stand pipe. At this point, all fuel will be used from the forward tank. Finally, after the forward tank is empty, the remaining fuel in the aft tank will be used.

Refer to paragraph 1.25.P.2 for operational information of the L/FUEL XFR and R/FUEL XFR LIGHTS and paragraph 1.25.R for information on the FUEL LOW LIGHT.

1.25.B Left Fuel Boost/XFR Alternate Electrical Circuit

In the event that a short circuit or battery hot condition occurs in the helicopter and all DC bus power is shutoff, it is desirable to maintain the operation of one transfer pump and one boost pump.

The BATT switch configures the DC power feed to the left fuel boost pump and the left fuel transfer pump between the DC bus and the helicopter battery (Figure 1-13).

In the event the battery switch is positioned to OFF during helicopter operations, an alternate circuit is provided to allow operation of the left fuel transfer and left fuel boost pumps. During this condition, with the fuel valve switch positioned to ON, battery voltage is supplied through the FUEL BOOST/XFR backup circuit breaker, the fuel valve switch, the battery switch, and the left fuel XFR/boost circuit breaker switch to the left fuel transfer and left fuel boost pumps.

1.25.C Fuel Systems Control

Fuel system controls are located on the instrument panel, overhead console, and aft lower right fuselage. The controls consist of a FUEL VALVE switch, a FUEL BOOST/XFR LEFT switch, a FUEL BOOST/XFR RIGHT switch, and two FUEL CELL DRAIN switches.

1.25.D Emergency Fuel Valve Switch

The EMERG FUEL VALVE switch is located on the lower right side of the instrument panel. It is a guarded two position toggle switch that provides a means of shutting off the flow of fuel to the engine.

The fuel shutoff valve is a motorized gate valve which must be driven to either the open or closed position. When activated by the EMERG FUEL VALVE switch, the shutoff valve motor will rotate in the appropriate direction to OPEN or CLOSE the gate valve. The EMERG FUEL VALVE light will momentarily illuminate during valve transit.

1.25.E Left and Right Fuel Boost/XFR Switch

The left and right FUEL BOOST/XFR switches (Figure 1-12) are a circuit breaker toggle design and located on the overhead console. Each switch controls the operation of its respective boost pump located in the main fuel tank and transfer pump in the forward fuel tank. In addition to the left and right FUEL/XFR switches, separate electrical circuits within the fuel signal conditioner are used to control the operation of the left and right fuel transfer pumps as the forward fuel tank is emptied.

The LEFT FUEL/XFR switch is outlined by a yellow border (Figure 1-5) to identify that it has an alternate circuit (Figure 1-16). In the event the BATT switch is positioned to OFF during helicopter operations, an alternate circuit is also provided to allow operation of the left fuel transfer and left fuel boost pumps. During this condition, with the EMERG FUEL VALVE switch positioned to ON, battery voltage is supplied through the FUEL BOOST/XFR BACKUP circuit breaker (located on the left side of the instrument pedestal above the chin bubble), the EMERG FUEL VALVE switch, the BATT SWITCH, and the LEFT FUEL BOOST/XFR circuit breaker switch to the left fuel transfer and left fuel boost pumps. Therefore, in this situation the LEFT BOOST pump will continue to run. The LEFT/XFR pump will continue to run as long as the fuel signal conditioner determines that there is fuel in the forward tank.

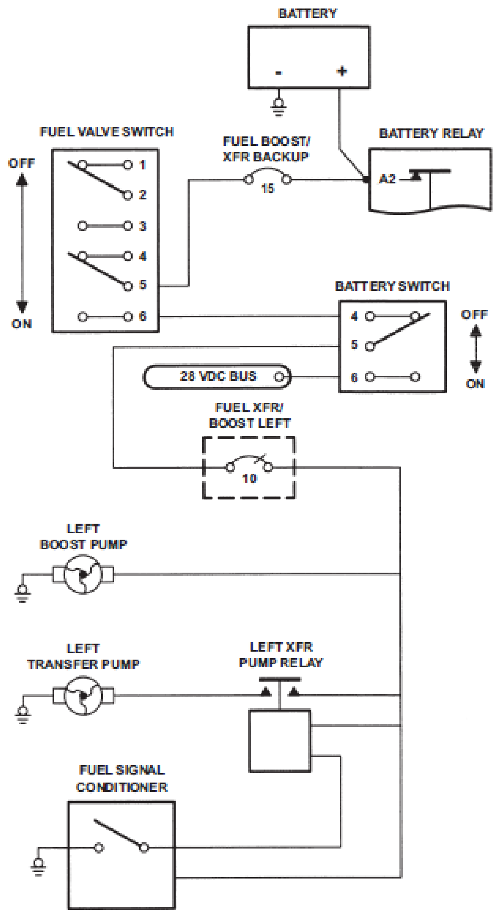


Figure 1-13 Left Fuel Boost/XFR Alternate Circuit Schematic

1.25.F Fuel Cell Drain Switches

The FUEL CELL DRAIN switches provide a means of draining fuel from both the forward and aft fuel cells individually. The switches are a momentary push button design made up of an outer environmental seal and an inner switch assembly. They are mounted side by side on the lower right side of the aft fuselage, below and aft of the fuel filler port.

The EMERG FUEL VALVE switch must be in the OFF position to operate either of the fuel cell drain valves. This prevents inadvertent operation of the fuel cell drains during flight.

With helicopter electrical power provided and the EMERG FUEL VALVE switch positioned to OFF, pressing the FUEL CELL DRAIN switches will OPEN the respective drain valves allowing fuel to be drained from the forward and main fuel cells.

1.25.G Fuel System Indicators

Fuel system indicators consist of a fuel quantity indicator, fuel pressure indicator, FUEL VALVE light, L/FUEL BOOST light, R/FUEL BOOST light, L/FUEL XFR light, R/ FUEL XFR light, FUEL FILTER light, and a FUEL LOW light.

1.25.H Fuel System Capacity

For fuel system capacity refer to the FMS-E407-789-1, Section 5.

1.25.I Fuel Quantity Gauging (FQGS)

The fuel quantity gauging system (FQGS) measures the quantity of fuel in the two main fuel tanks. The FQGS also measures the quantity of fuel in the auxiliary fuel tank when it is installed.

The fuel quantity is measured by three capacitance type probes in the fuel tanks. The signals from the probes are used by the fuel signal conditioner to calculate the fuel weight. The signal conditioner provides a signal to the fuel quantity gauge to display the calculated fuel weight.

1.25.J Fuel Quantity Signal Conditioner

The signal conditioner is a separate unit that is located on the aft electrical equipment shelf next to the DC controller (Voltage Regulator).

The electronic interface circuits for the fuel low level detection system are located along with fuel gauging system signal conditioner circuits in the same physical unit.

Both systems are physically and electrically separate within the unit. There are three LEDs located on the aft side of the signal conditioner which are used to display the FQGS status for purposes of troubleshooting.

1.25.K Signal Conditioner Built-In-Test (BIT)

1.25.K.1 Power-up BIT

The signal conditioner receives power from the 28 VDC bus through the FUEL QTY INSTR circuit breaker. The signal conditioner carries out a power up BIT when the unit is first provided power. The power-up BIT must be completed before the signal conditioner can take any readings of fuel quantity. The signal conditioner should complete a power-up BIT check within approximately 4 seconds after application of power. There is no connection between the BIT feature of the signal conditioner and the BIT performed by the fuel quantity gauge.

If a failure is detected during the power-up BIT or if an error is found in the probe signal received or the power source for the probe input, the signal conditioner will blank the fuel quantity gauge.

If errors have been detected and the gauge has been blanked, the signal conditioner will not turn the gauge back on even if the error has been corrected unless the power is turned OFF and ON to the signal conditioner.

In addition, the failures detected will be displayed on three LEDs on the back of the signal conditioner (refer to the ICA-E407-789 for fault explanation).

1.25.K.2 Continuous BIT

The signal conditioner carries out a continuous BIT whenever it is powered. If a failure is detected during the continuous BIT or if an error is found in the probe signal received or the power source for the probe input, the signal conditioner will blank the gauge display.

If errors have been detected and the gauge has been blanked, the signal conditioner will not turn the gauge back on even if the error has been corrected un less the power is turned OFF and ON to the signal conditioner.

In addition, the failures detected will be displayed on three LEDs on the back of the signal conditioner (refer to the ICA-E407-789 for fault explanation).

1.25.L Fuel Quantity Calculation

A microprocessor in the signal conditioner uses the information provided by the three fuel probes to compute the weight of the fuel in the following steps.

1. Uses the Main Tank Forward Fuel Probe (Probe No. 2) input signal for density correction if it is totally immersed in fuel or if not, uses a default density (default density is 6.6594 pounds/gallons).
2. Calculates the height of the fuel indicated for each of the three probes corrected for fuel density by using the value from step 1.
3. The calculated height on each probe is used to look up a volume of fuel in gallons in a table contained in the NVM of the signal conditioner.
4. The weight of the fuel is then calculated by multiplying the volume by the density. A calculation is done for all three probes to compute the total system weight of fuel. A calculation is also done for only the Forward Tank Fuel Probe (Probe No. 3).

Total system weight information is transmitted from the signal conditioner to the fuel quantity gauge. When the forward fuel quantity switch is pushed, the weight for the forward tank only is transmitted.

1.25.M Fuel Quantity Gauge

The usable fuel weight (in pounds) is calculated by the fuel signal conditioner and is displayed by the gauge.

1.25.N Fuel Quantity Button

The fuel quantity gauge normally indicates the total usable fuel in both fuel tanks and auxiliary tank (if installed). Pushing the FUEL QTY FWD TANK button will make the fuel quantity gauge display the fuel in the forward tank only.

1.25.O Fuel Pressure/Ammeter Gauge

The fuel pressure/ammeter gauge is a dual display gauge. The left side of the instrument displays the pressure output from the two fuel boost pumps in pounds per square inch (PSI). The instrument receives its input signal from a transducer mounted between the fuel boost pumps and the fuel shutoff valve. The minimum pressure limit is set to ensure that sufficient fuel pressure will be supplied to the input of the engine driven fuel pump.

1.25.P Fuel Valve Light

The FUEL VALVE light will illuminate when the emergency fuel shut off valve is in transit or has stopped somewhere between the full OPEN or full CLOSED position.

1.25.P.1 L/Fuel and R/Fuel Boost Lights

The L/FUEL BOOST and R/FUEL BOOST lights will illuminate when their respective fuel pressure switch senses a decreasing boost pump output pressure of 1.5 ±0.5 PSI. When boost pump pressure is increasing, the lights will extinguish prior to the pressure passing through 5 PSI.

1.25.P.2 L/Fuel XFR and R/Fuel XFR Lights

1.25.P.2.A S/N 53000 Through 53174

The L/FUEL XFR and R/FUEL XFR lights (Figure 1-12, Sheet 1) will illuminate when their respective fuel pressure switch senses a decreasing boost pump output pressure of 1.5 ±0.5 PSI. When boost pump pressure is increasing the lights will extinguish prior to the pressure passing through 5 PSI.

Fuel must be present in the forward fuel cell for the annunciator circuits to operate since they are both controlled by the fuel signal conditioner. When the forward fuel cell is nearing depletion the signal conditioner uses time delays to control the transfer pumps and lights. The time delay allows continued operation of the transfer pumps to ensure all of the fuel in the forward tank is transferred to the main fuel tank. The forward fuel cell will be empty when approximately 193.1 pounds of total fuel is indicated.

The fuel signal conditioner time delay which controls the L/FUEL XFR pump circuit is provided by the FUEL XFR/BOOST LEFT circuit breaker. The fuel signal conditioner time delay which controls the R/FUEL XFR pump circuit is provided by the FUEL XFR/ BOOST RIGHT circuit breaker.

The L/FUEL XFR and R/FUEL XFR light circuits will remain operational during the time delay and will illuminate in the event transfer pump output pressure drops below 1.5 ±0.5 PSI.

Once the time delay periods are ended, the transfer pumps and the L/FUEL XFR and R/FUEL XFR light circuits are deactivated. The transfer pump and light circuits will stay inoperative as long as the forward fuel cell is empty. Reactivation of the transfer pump circuits will occur when approximately 18 pounds of fuel enters the forward fuel cell. As the forward fuel cell is empty at approximately 193.1 pounds of fuel, if the helicopter is shutdown at, or being refueled to between 193.1 and approximately 211.1 pounds of total fuel, it is possible that up to 18 pounds of fuel may remain in the forward fuel cell as unusable.

1.25.P.2.B S/N 53175 and Subsequent

Activation of either annunciator circuit is dependent on the relationship between the fuel quantity in the forward fuel tank and the total quantity of the fuel system.

The L/FUEL XFR or R/FUEL XFR transfer light (Figure 1-15, Sheet 2) will illuminate when their respective fuel pressure switch senses a decreasing boost pump pressure of 1.5 ±0.5 PSI, provided the conditions shown in Table 1-8 (Transfer Light Activation Table) are met:

Table 1-8 Transfer Light Activation Table, S/N 53175 and Subsequent

FWD Tank <25 Lbs for 10 Consecutive Seconds	Total Fuel <250 Lbs for 10 Consecutive Seconds	L/Fuel or R/Fuel XFR Light Illuminated (At A Decreasing Pressure of 1.5+/-0.5 PSI)
FALSE	FALSE	YES
FALSE	TRUE	YES
TRUE	FALSE	YES
TRUE	TRUE	NO

With a L/FUEL XFR or R/FUEL XFR pump pressure of 5 PSI or greater, the respective left and right fuel pressure switches will open and cause the L/FUEL XFR or R/FUEL XFR lights to go off.

When the fuel signal conditioner detects a forward fuel tank quantity of less than 25 pounds and a total fuel system quantity of less than 250 pounds, for 10 consecutive seconds, L/FUEL XFR or R/FUEL XFR transfer lights will be deactivated to prevent intermittent light flickering while the remaining fuel is transferred from the forward fuel tank to the aft fuel tank.

When forward fuel tank depletion is detected by the number 3 fuel probe and the low level detector, the input signals to the fuel signal conditioner are removed. With the input signals removed, the fuel signal conditioner utilizes two 360 second time delays prior to removing the ground to the right and left transfer pump relays. This allows the right and left transfer pumps to continue running for 360 seconds to ensure all the fuel in the forward tank is transferred to the main fuel tank. The forward fuel cell will be empty when approximately 193.1 pounds of total fuel is indicated.

The annunciator and transfer pump circuits will stay inoperative until the fuel system is refueled with an appropriate amount of fuel to reactivate the system. Reactivation of the transfer pump circuits will occur when approximately 18 pounds of fuel enters the forward fuel cell. As the forward fuel cell is empty at approximately 193.1 pounds of fuel, if the helicopter is shut down at, or being refueled to between 193.1 and approximately 211.1 pounds of total fuel, it is possible that up to 18 pounds of fuel may remain in the forward fuel cell as unusable.

1.25.Q Airframe Fuel Filter Light

The A/F FUEL FILTER light will illuminate if the airframe fuel filter is in an impending bypass condition. This will occur when a differential pressure of 0.875 ±0.125 PSI is present between the input and output of the fuel filter. The airframe fuel filter will go into bypass at 3.75 ±0.25 PSI.

1.25.R Fuel Low Light

The FUEL LOW light circuit is designed to alert the pilot that the main fuel tank quantity is low. Illumination will occur when approximately 100 ±10 pounds of usable fuel remains in the main fuel cell.

A float type low level switch is used to detect the fuel low condition. The input from the low level detector is passed through the fuel signal conditioner which provides a 13 ±3 second time delay to reduce the possibility of intermittent annunciator flickering due to fuel sloshing.

In addition, momentary activation (less than 2 seconds) of the FUEL LOW light may be visible when power is applied to the helicopter and fuel quantity is greater than the required low level activation point. This is a normal occurrence and is a function of fuel signal conditioner power up logic.

1.26 Retirement Index Number (RIN)

Each component with a retirement life sensitive to "TORQUE EVENTS" will be assigned a maximum RETIREMENT INDEX NUMBER (RIN). This RIN corresponds to the maximum allowed fatigue damage resulting from lifts and takeoffs. A new component will begin with an accumulated RIN of zero that will be increased as lifts and takeoffs are performed. The operator will record the number of lifts and takeoffs and increase the accumulated RIN accordingly. When the maximum RIN is reached, the component will be removed from service. Certain components may be assigned a life in hours in addition to the RIN.

Pilots are to record “TORQUE EVENTS” for each flight. Maintenance personnel will convert the “TORQUE EVENTS” into “RETIREMENT INDEX NUMBERS” (RIN) to track the lives of all required components.

A "TORQUE EVENT" is defined as a takeoff (one takeoff plus the subsequent landing = one RIN) or a lift (internal or external). For example, if an operator performs six takeoffs and ten sling loads, this would total 16 torque events: (6 takeoffs = 6 events, 10 sling loads = 10 events, 6 + 10 = 16 events total).

1.27 Transmission

The transmission and mast assembly (Figure 1-17) transfers the engine torque to the main rotor system with a two stage gear reduction of 15.29 to 1.0 (6317 to 413 RPM).

The transmission assembly is made up of a top support case and lower case which contains an input pinion and bevel gear arrangement, a planetary gear train, and an accessory gear drive. The components that are attached to the transmission and mast assembly are the engine to transmission driveshaft, transmission oil pump, transmission oil filter housing, hydraulic pump, rotor RPM monopole pickup, and two electric chip detectors.

The transmission assembly is attached to the roof of the helicopter, forward of the engine by a pylon installation. The pylon installation uses two side beams, four elastomeric corner mounts, and two for/aft restraint springs.

1.27.A Deleted

1.27.B Transmission Oil System

The transmission oil system (Figure 1-15) lubricates the transmission and the mast. An oil level sight gauge is located on the right side of the transmission lower case and may be viewed through a cutout in the air induction cowling. A non- vented filler cap is located on the right side of the top case to fill the transmission oil system.

The transmission accessory gear drives the oil pump which delivers 6.0 to 6.7 GPM. The pressure is controlled by a pressure regulator valve which is set at approximately 52 PSI. The pump scavenges oil from the lower case sump through a wire screen and the lower chip detector. It is then directed to the transmission mounted oil manifold and the filter element.

To ensure oil flow is not restricted, the filter incorporates an impending bypass indicator button on the end of the filter housing which will extend at 14 ±2 PSI. Ensure appropriate maintenance actions are carried out following an impending bypass indication in accordance with the ICA-E407-789. Additionally; a filter bypass valve will open if the differential pressure reaches 17 PSI and if the differential pressure reaches 29.6 to 38.6 PSI, a high pressure relief valve will open and the oil will totally bypass the filter.

The transmission mounted oil manifold also incorporates a thermostatic valve which controls the flow of oil to the oil cooler. At oil temperatures below 150°F (66°C), the oil cooler is bypassed and the oil is directed back to the transmission. As the oil temperature increases above 150°F (66°C) oil is gradually directed to the oil cooler until at 180°F (82°C), all oil is directed through the cooler.

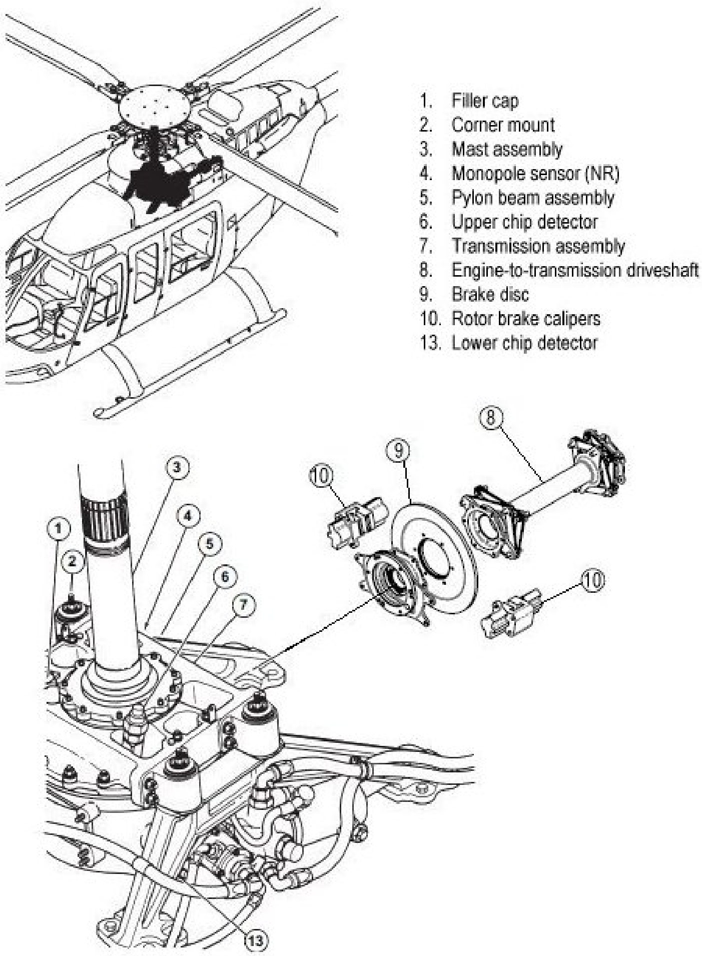


Figure 1-14 Transmission Assembly

A temperature bulb for the oil temperature indicator and a thermoswitch for the XMSN OIL TEMP light are also located on the transmission mounted oil manifold.

After the oil exits the cooler (or bypasses the cooler through the thermostatic bypass valve), it is then directed to the transmission to lubricate the various gears and bearings. The oil is then directed to a deck mounted oil manifold located below the transmission input driveshaft. A pressure transducer and oil pressure switch are mounted on the transmission deck oil manifold. The transducer provides signals to the oil pressure gauge and the pressure switch controls the XMSN OIL PRESS light.

1.27.C Transmission Indicators

Transmission indicators include an oil temperature and pressure gauge, XMSN OIL TEMP light, XMSN OIL PRESS light, and XMSN CHIP light.

1.27.D Transmission Oil Temperature and Pressure Gauge

The transmission oil temperature and pressure gauge is a dual instrument that simultaneously displays oil temperature in degrees Celsius on the right side display and oil pressure in PSI on the left side display. Each side of the indicator is powered by its own circuit breaker.

The transmission temperature input signal is provided by a thermo bulb installed on the transmission oil filter manifold. The transmission oil pressure is provided by a transducer mounted on the transmission deck oil manifold.

1.27.D.1 Transmission Oil Temperature Gauge

The XMSN OIL TEMP light will be illuminated when the transmission oil temperature switch detects a temperature of 110 ±5.6°C (230 ±10°F). The oil temperature switch is mounted on the transmission oil filter manifold housing.

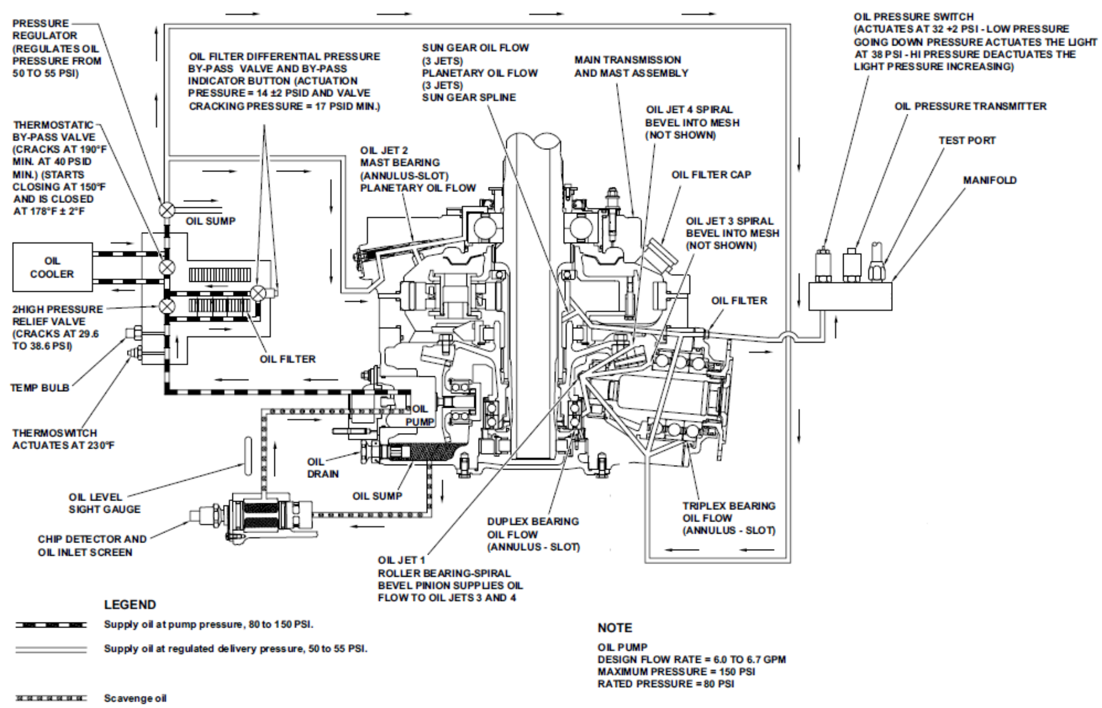


Figure 1-15 Transmission Oil System

1.27.D.2 Transmission Oil Pressure Light

The XMSN OIL PRESS light will be illuminated when the transmission oil pressure switch detects a decreasing oil pressure of 30 ±2 PSI.

Similarly, as the transmission oil pressure builds, the oil pressure switch will extinguish the XMSN OIL PRESS annunciator prior to the pressure passing through 38 PSI.

1.27.E Transmission Chip Annunciator

The XMSN CHIP light will illuminate if magnetic particles in the oil system accumulate on any one of the two quick disconnect magnetic chip detectors.

Two magnetic chip detectors, one on the upper case and one on the lower case, are mounted on the main transmission. The lower case chip detector incorporates a self-sealing valve which prevents the loss of oil from the gearbox when the chip detector is removed.

1.28 Rotor System

1.28.A Main Rotor Hub and Blades

The rotor assembly (Figure 1-16) is a four bladed soft-in-plane design with a 35 foot diameter rotor.

The main rotor hub contains a glass/epoxy composite yoke that acts as a flapping flexure. Elastomeric bearings and dampers which require no lubrication are utilized. The hub also incorporates the use of lead-lag, coning/flapping and droop stops.

The main rotor blades are a composite design utilizing a glass/epoxy spar, glass/epoxy skins, and a nomex core after-body. The blades incorporate a nickel plated stainless steel leading edge erosion strip and are coated with conductive paint for lightning protection. The blades are also individually interchangeable.

1.28.B Tail Rotor Hub and Blades

The tail rotor (Figure 1-17) is a two-bladed teetering rotor with a 5.42 foot diameter. It is mounted on the left side of the tailboom and rotates clockwise when looking inboard from the left side of the helicopter.

Teflon lined pitch change bearings are installed in a steel yoke assembly which utilizes an elastomeric flapping bearing and flapping stops.

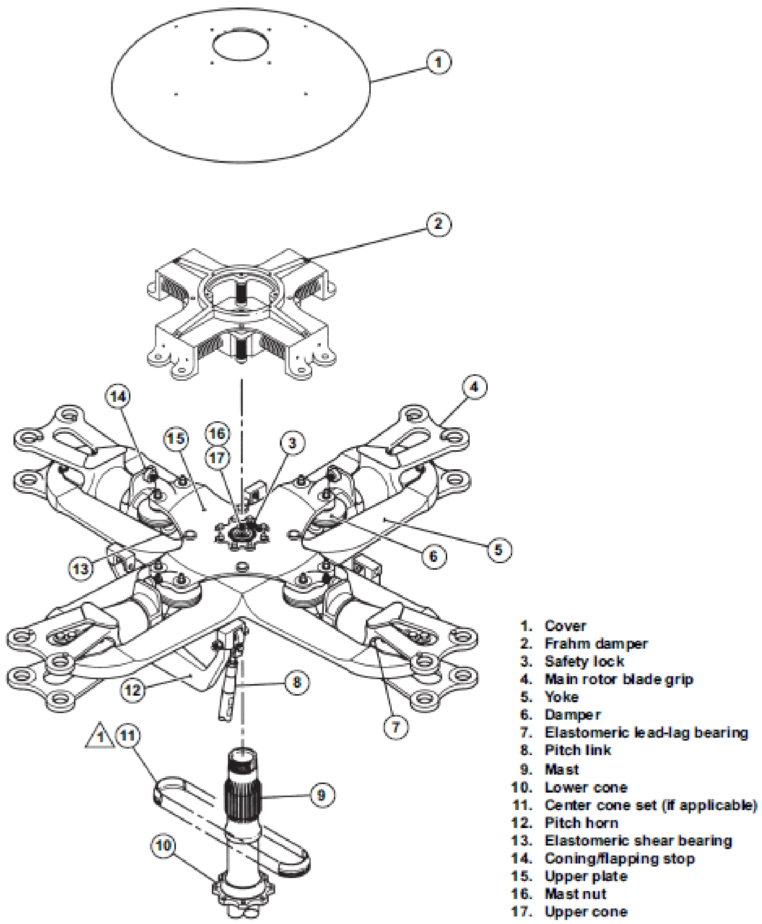
The blades are a composite design utilizing a glass/epoxy spar, glass/epoxy skins, and a nomex core. The blades incorporate nickel plated stainless steel leading edge abrasion strip and are coated with conductive paint for lightning protection.

The tail rotor yoke static stop has been designed with yield indicators. The yield indicators provide the ability to visually determine if the tail rotor yoke has been stressed beyond designed limits. This will be evident by deformation of either of the static stop yield indicators due to excessive contact with the yoke (Figure 1-17). If deformation of either yield indicator is evident, contact maintenance personnel prior to further flight.

1.28.C Tail Rotor Gearbox

The tail rotor gearbox (Figure 1-17), located on the aft end of the tailboom, drives the tail rotor. It contains two spiral bevel gears positioned at 90° angles to the other. The tail rotor gear box has a gear reduction of 2.53 to 1.0 which reduces the driveshaft input speed of 6317 RPM to an output shaft speed of 2500 RPM.

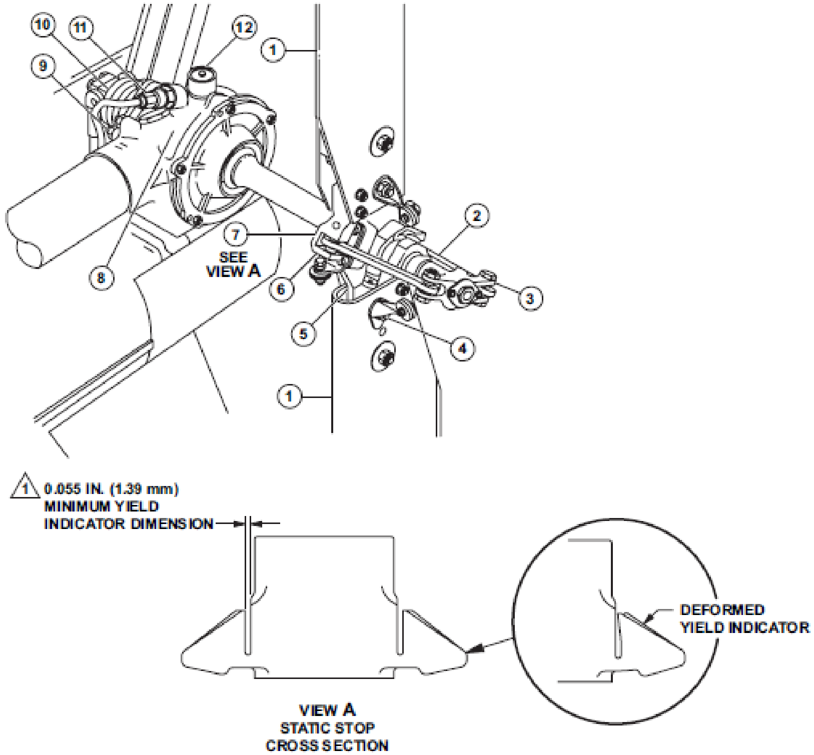
The gearbox has a self-contained oil lubrication system, non-vented filler cap, and a magnetic chip detector.



NOTE

1. Center cone set applicable to S/N 53000 through 53631 Pre TB 407-05-66. S/N 53632 and subsequent have an integral mast center cone.

Figure 1-16 Main Rotor Assembly



1. Tail rotor blade
2. Tail rotor pitch change link
3. Crosshead
4. Counterweight assembly
5. Tail rotor hub
6. Pitch horn
7. Static stop
8. Tail rotor gearbox
9. Chip detector
10. Pitch change mechanism
11. Breather line
12. Tail rotor gearbox filler cap

NOTE

1 S/N 53351 and subsequent, and helicopters modified per ASB 407-99-27 or TB 407-99-17.

Figure 1-17 Tail Rotor Assembly

1.28.D Tail Rotor Chip Light

The T/R CHIP light will illuminate if magnetic particles in the oil accumulate on the magnetic chip detector.

The chip detector incorporates a self-sealing valve which prevents the loss of oil from the gearbox when the chip detector is removed.

1.29 Rotor System Indicators

1.29.A Dual Tach Gauge

The Dual Tachometer has two pointers that simultaneously display Rotor RPM (NR) on the outer scale and the engine NP on the inner scale. All scales are in percent RPM.

Three gearbox-mounted sensors are used to measure the rotational speed of the power turbine spool. Each ECU channel has a dedicated coil for the NP signal and one is shared between the channels.

An NR speed pick up mounted on the transmission lower case provides three separate identical signal outputs for rotor RPM. Each FADEC/ECU channel receives a signal from two of the outputs and the other signal is sent to Rotor RPM sensor switch.

1.29.B RPM Light and Warning Horn

The RPM light and warning horn circuit is designed to activate when the main rotor RPM (NR) is less than 95%. The RPM light will also be illuminated at a NR speed of 107% and higher.

1.29.B.1 NR Less Than 95%

If the rotor RPM sensor detects a main rotor RPM (NR) of less than 95%, it will illuminate the RPM light and (continuous sounding) low rotor RPM horn. The low rotor RPM horn can be muted by pressing the warning horn mute switch.

1.29.B.2 NR 107% Or Greater

If the rotor RPM sensor detects a main rotor RPM NR of 107% or greater, it will illuminate the RPM light. The low rotor RPM horn will not be activated.

1.30 Flight Control System

1.30.A Rotor Controls

Main rotor and tail rotor flight control systems (Figure 1-18), consisting of cyclic, collective and anti-torque controls are used to regulate the helicopter attitude, altitude and direction of flight. The flight controls are hydraulically boosted to reduce pilot effort and to counteract control feedback forces.

1.30.B Main Rotor

Main rotor cyclic and collective flight controls regulate pitch and roll attitude and thrust. Control inputs from the cyclic and collective control sticks (Figure 1-18) in the cockpit are transmitted by push-pull tubes to hydraulic servo actuators mounted on the top deck. The actuators operate the cyclic and collective levers, which raise, lower, and tilt the swashplate. The swashplate converts fixed control inputs to the rotating controls and allows cyclic and collective pitch inputs to the main rotor.

In the case of loss of hydraulic pressure to the servo actuators, springs are installed in parallel to the cyclic and collective push-pull tubes on the cabin roof to assist the pilot with the increased control feedback forces.

1.30.B.1 Cyclic

The cyclic control stick (Figure 1-18) is mounted under the pilot’s crew seat and protrudes from the forward bulkhead of the crew seat. The fore and aft cyclic input is connected through push-pull tubes to the cyclic hydraulic servo actuators. In addition, all cyclic fore and aft movement is fed through a cam assembly that automatically adds an amount of lateral cyclic input that is a percentage of the fore and aft cyclic movement. A spring canister is provided in line with the cam input to permit cyclic movement in the event that the cam

assembly becomes jammed.
The lateral cyclic input is connected through push pull tubes to the cyclic hydraulic servo actuator. The hydraulic servo actuators operate bellcranks and push pull tubes that tilt the swashplate non-rotating ring. The swashplate rotating ring tilts likewise and actuates the pitch links which control the plane of rotation of the main rotor.

The cyclic control stick grip contains a two- position intercommunication/radio transmit switch and a cargo hook release switch. An adjustable friction control knob, located at the base of the cyclic stick where it protrudes through the forward crew seat bulkhead, allows the pilot to set the desired amount of control stiffness for flight or to lock the cyclic control stick during ground operation or shutdown.

A cyclic stick position switch (cyclic centering switch) is attached to the cyclic stick bellcrank. When the helicopter is on the ground, this switch will cause the CYCLIC CENTERING annunciator in the caution/ warning panel to illuminate when the stick is not centered.

1.30.B.2 Collective

The collective control stick (Figure 1-18) is mounted between the pilot and copilot crew seats. The collective control stick controls the collective hydraulic servo actuator through push-pull tubes. This operates the collective lever mounted on the top of the transmission. The collective lever raises and lowers the swashplate ball-sleeve assembly and the cyclic levers to induce collective pitch to the main rotor blades without affecting the cyclic path. A spring is installed under the copilot's crew seat to balance the required force to raise and lower the collective with the hydraulic boost system operating.

A collective friction knob is located near the base of the collective stick between the pilot and copilot seats. A throttle twist grip for the engine is mounted on the collective stick. A mechanical idle release push button is located in front of the twist grip throttle. A switchbox located on the forward end of the collective stick provides a base for the engine start switch and landing light switch.

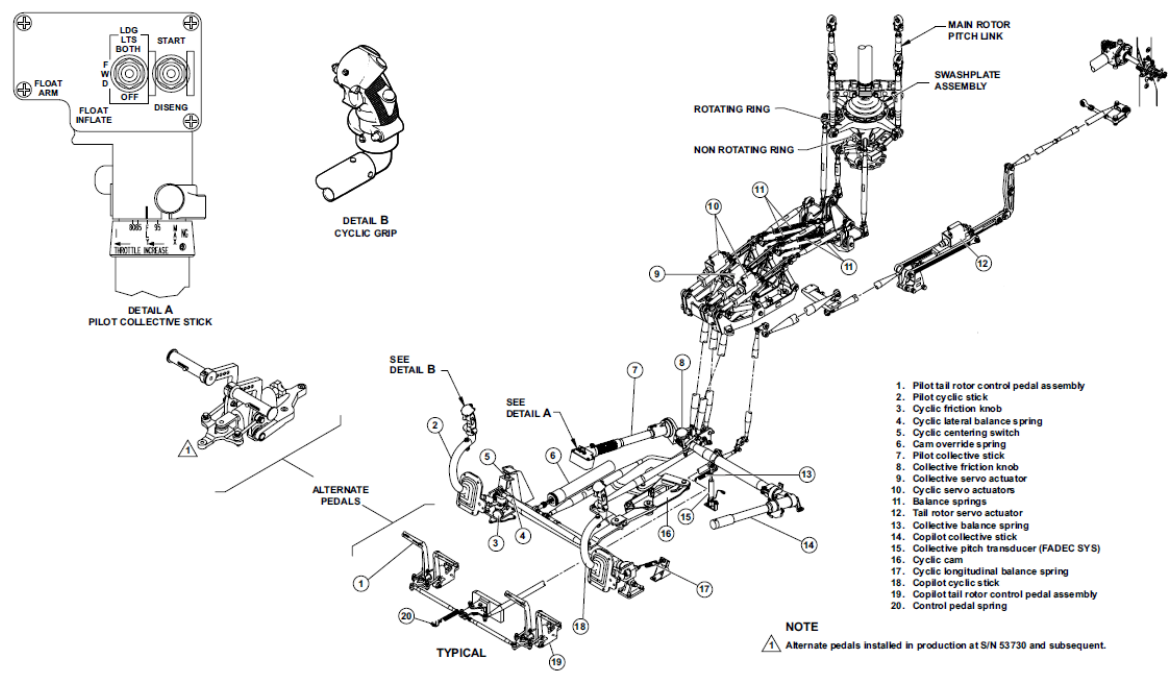


Figure 1-18 Flight Controls

1.30.C Tail Rotor

The tail rotor, or anti-torque, flight controls (Figure 1-18) provide pitch adjustment of the tail rotor blades for yaw control. A set of pedals on the cockpit floor, forward of the pilot seat, are connected to a directional control hydraulic servo actuator, located in the aft fuselage near the tailboom. Push-pull tubes connect the actuator to the fixed pitch change mechanism of the tail rotor gearbox. The tail rotor fixed mechanism is connected to the rotating controls through a rotating push-pull tube. The push-pull tube attaches to a sliding crosshead that moves in and out on splines on the tail rotor mast to provide pitch control. Rotating counterweights minimize the control forces required.

The tail rotor control pedals contain a bellcrank pedal adjuster, which provides for manual adjustment of pedal position according to the pilot's needs. Alternate pedals may also be installed which allow the pilot to manually adjust the position of the pedal foot rests.

For helicopters with dual controls, the copilot fully functional tail rotor control pedal assembly is installed on the floor in front of the copilot seat to provide a means for the copilot to control the tail rotor assembly. The control pedals are linked to the pilot pedals by means of control tubes and a bellcrank. The copilot pedals can also be positioned, as desired, by means of the pedal adjuster. Alternate pedals may also be installed which allow the copilot to manually adjust the position of the pedal foot rests.

1.30.D Airspeed Actuated Pedal Stop System

The Airspeed Actuated Pedal Stop system is comprised of a Pedal Restrictor Control Unit (PRCU), an actuating rotary solenoid, positioning sensing micro switch, and a press-to-test PEDAL STOP (PTT) switch.

A PEDAL STOP segment (Figure 1-4) on the Caution/Warning panel indicates any malfunction of the system. System power is provided through a 5-amp circuit breaker located in overhead console (Figure 1-5).

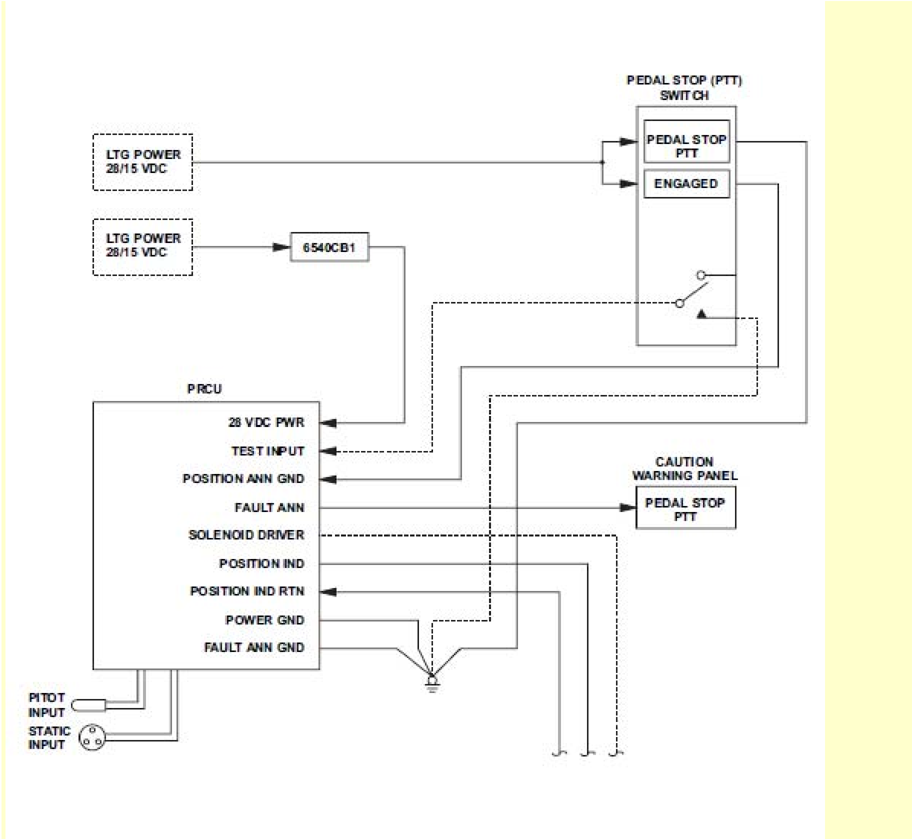


Figure 1-20 Airspeed Actuated Pedal Stop System

The helicopter pitot and static installation interfaces with the PRCU. The PRCU calculates airspeed from the pitot and static inputs and when greater than 55 ±5 KIAS, drives the solenoid to extend the pedal stop restrictor into the left pedals range of travel. The PRCU will trigger the solenoid to retract the pedal stop when calculated airspeed falls below 50 ±5 KIAS.

Upon full extension of the pedal stop, the position sensing microswitch is activated and the PRCU illuminates the ENGAGED message on the PEDAL STOP (PTT) switch. This message is extinguished when the pedal stop is retracted.

The Airspeed Actuated Pedal Stop System diagram is shown in Figure 1-19.

1.31 Hydraulic System

The hydraulic system (Figure 1-20) provides boost power for the cyclic, collective and anti-torque flight controls. The system includes a pump, reservoir, pressure and return filter assemblies, pressure and return manifold, pressure monitoring sensor, solenoid valve, pressure relief valve, flight control servo actuators, and interconnecting tubing and fittings.

The hydraulic pump is mounted on and driven by the transmission. The pump is a variable delivery pressure compensated self-lubricated type designed to operate continuously and provide a rated discharge pressure of 1000 -25/+50 PSI. Hydraulic fluid is supplied to the pump from a vented gravity feed reservoir mounted forward of the transmission. Fluid passes through the pump, a pressure filter, and a solenoid valve. The solenoid valve is controlled by a HYD SYS switch located on the overhead console. When HYD SYS switch is ON, the solenoid valve is de-energized to the open position and fluid is routed to cyclic, collective, and anti-torque actuators. From the actuators, fluid passes through a return filter to the reservoir. Both the pressure and return filter assemblies have filter indicators. These indicators are activated when the differential pressure across the filter is 70 ±10 PSI. The pressure filter does not contain a bypass valve. The pressure filter will clog completely in order to prevent contaminated hydraulic fluid from being pumped through the system. The return filter contains a bypass valve that will allow fluid to bypass the filter if it senses a differential pressure of

100 ±25 PSI across the filter. When the differential pressure decreases to 60 PSI, the bypass valve will close. This allows fluid returning from the servo actuators to return to the reservoir even if the return filter is completely clogged.

A relief valve is incorporated in the system, between the pressure filter and the solenoid valve. The relief valve is normally closed. When the pressure reaches 1225 ±150 PSI, the relief valve will open to protect the system from damage. The relief valve will reset when pressure drops to approximately 1075 PSI.

1.31.A Hydraulic Indicators

Hydraulic system indicators include a HYDRAULIC SYSTEM caution light on the caution/warning panel and filter bypass indicators located on both the pressure and return filter assemblies.

1.31.B Hydraulic Filter Indicators

Each filter assembly contains a filter indicator that indicates an impending clogged filter. The indicator consists of a red button mounted on the filter assembly housing. When the differential pressure across the filter is 70 ±10 PSI, the red button will rise. To prevent inaccurate indications of bypass, the indicator will not work when the hydraulic fluid temperature is less than 35°F (2°C). If hydraulic fluid temperature is more than 35°F (2° C), the indicator gives the correct indication of clogging, even if the ambient temperature is below 35°F.

If a filter button is extended, indicating an impending clogged filter, maintenance action should be performed prior to next flight. The filter indicator is reset by pushing the button in

1.31.C Hydraulic System Light

The HYDRAULIC SYSTEM light circuit is designed to illuminate when the hydraulic system fluid pressure is below the minimum limit.

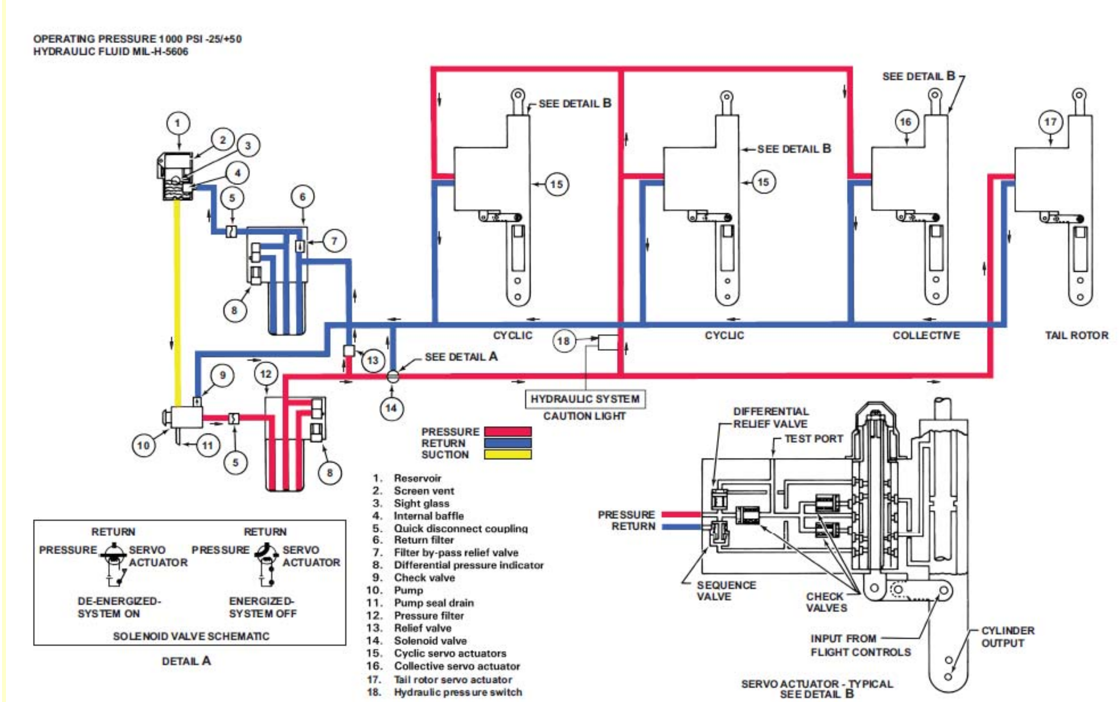


Figure 1-21 Hydraulic System

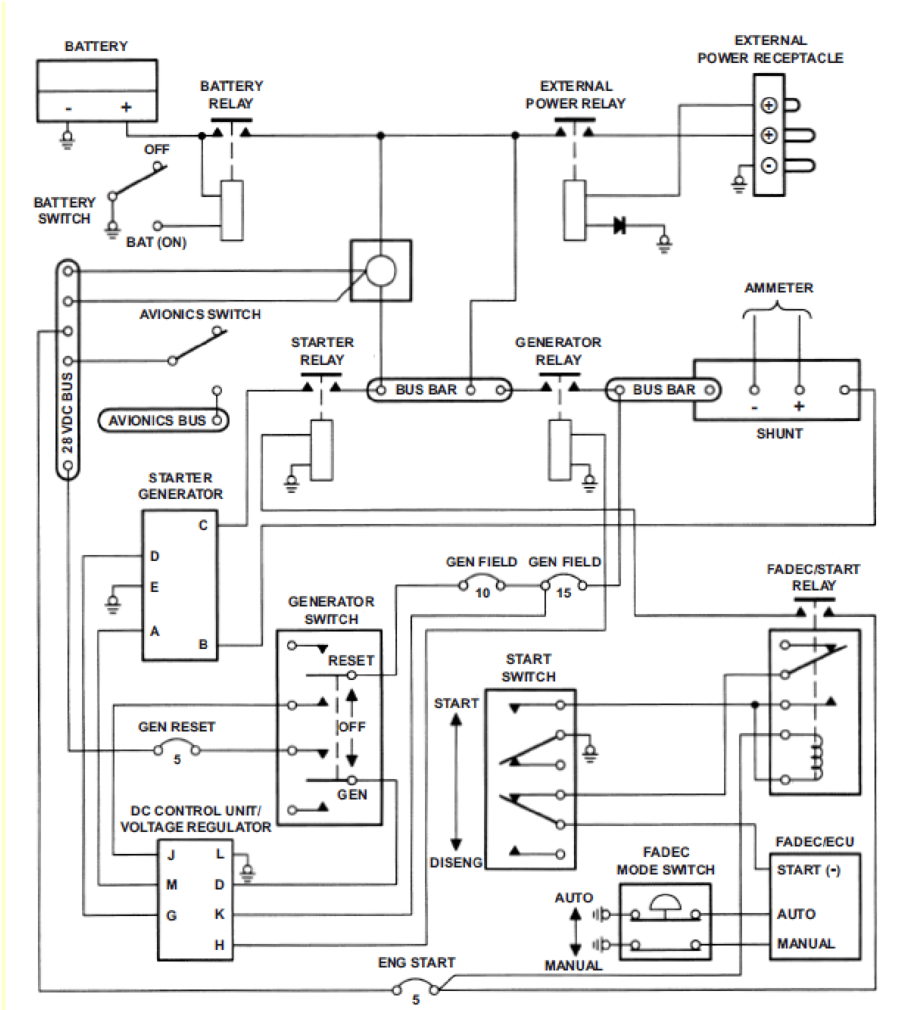


Figure 1-22 DC Electrical System

The hydraulic system light will be illuminated when the hydraulic pressure switch detects a decreasing pressure of 650 -0/+100 PSI, and will extinguish on an increasing pressure at 750 +0/-100 PSI. The hydraulic pressure switch is mounted on the hydraulic manifold, forward of the hydraulic actuator support.

1.32 Electrical System

The helicopter is equipped with a 28 VDC electrical system (Figure 1-21). Power for this system is obtained from a 24 volt, 34 amp/hour battery and a 30 volt, 200-amp starter-generator. The starter-generator has been derated to 180 amps to ensure adequate cooling under all operating conditions up to 18,000 feet Hp. Refer to the FMS-E407-789 limitations for operations above 18,000 feet Hp.

Major components of DC power system include battery, starter-generator, DC control unit, voltage regulator, relays, 28 VDC bus, and circuit breakers. All circuits in electrical system are single wire with fuselage common ground return. Negative terminals of starter-generator and battery are grounded to helicopter structure. Controls for electrical system are located on overhead console and instrument panel.

Generator is provided with over voltage, under voltage and reverse current protection. If an over voltage (32 ±0.5 VDC), under voltage (18 ±1.8 VDC), or reverse current (0.08 to 0.150 VDC for 350 milliseconds) is detected, the DC control unit/voltage regulator will disconnect the generator from the system. Failure of the generator can be determined by GEN FAIL light, a zero ammeter reading, and battery voltage displayed on the voltmeter.

In the event power from the generator is lost, emergency power available from the battery can be maximized by pulling the circuit breakers on all non-essential systems. In the event of a total electrical failure (hard short on bus), if the battery switch is immediately placed to the OFF position, the 34 amp/hour battery, assuming 80% charged, can supply fuel boost and transfer pumps for a period of approximately 3.4 hours.

1.32.A External Power

External power maybe supplied to the helicopter by means of a receptacle located on the lower front section of helicopter. 28 VDC Ground Power Unit (GPU) shall be 500 amps or less to reduce risk of starter damage from overheating.

If external power was used to power the start and the battery switch was left in the OFF position, it is important to position the battery switch to ON prior to removing the external power source (refer to FMS-E407-789-1, Normal Procedures). If all sources of electrical power are removed from the ECU with the engine at idle in AUTO mode, the start solenoid valve in the HMU will open, causing the engine to decelerate and possibly flame out. If the Battery switch is inadvertently left OFF and the external power source is removed, do not attempt to reapply power when a decrease in NG speed is noted. Throttle should be positioned to OFF. Reapplication of electrical power could cause an over temperature condition due to the reduced NG speed and reintroduction of fuel by the FADEC system.

1.32.B Battery Switch

The battery switch is installed on the overhead console and controls the battery relay which connects the battery to the DC bus. The switch has two positions OFF and BATT.

If BATT switch is turned OFF and BATTERY RLY light illuminates, this indicates battery relay contacts have not opened. If battery relay contacts have not opened, battery will continue to receive a charge from the generator. To prevent this, pilot should turn GEN switch to OFF. Battery will continue to run all electrical systems through the closed battery relay. To reduce load, pilot should pull circuit breakers on all non-essential systems. Left FUEL XFR/ BOOST circuit breaker switch should be left ON to insure fuel transfer from the forward fuel tank to the main fuel tank continues.

The BATT switch also configures the DC power feed to the left fuel boost pump and the left fuel transfer pump between the DC bus (battery switch positioned to BATT) and the helicopter battery (BATT switch positioned to

OFF).

In the event the battery switch is positioned to OFF during helicopter operations, an alternate circuit (Figure 1-13) is provided to allow operation of the left fuel transfer and left fuel boost pumps. During this condition, with the EMERG FUEL VALVE switch positioned to ON, battery voltage is supplied through the fuel boost/xfr backup circuit breaker, the fuel valve switch, the battery switch, and the left fuel xfr/ boost circuit breaker switch to the left fuel transfer and left fuel boost pumps.

1.32.C Battery Charging

As a maintenance function, battery charging may be accomplished with battery installed in helicopter, due to minor depletion, using a GPU. The GPU must incorporate a good quality constant voltage regulator, a variable voltage selector and an amperage indicator.

Battery switch may be set to ON after GPU power is applied and voltage adjusted to 28.5 VDC (do not exceed 28.5 volts). Battery charging is supplied by the GPU and must be monitored. Charging will be completed when GPU output indicates approximately 8 amps.

1.32.D Generator Switch

The generator switch is installed on the overhead console and controls generator output by opening and closing the generator field circuit. The switch is a double pole, double throw, spring loaded design with only momentary contact in the RESET position. The switch has 3 positions, GEN, OFF, and RESET.

With the generator switch positioned to GEN, its function is to complete the generator field circuit between the starter generator and the generator control unit/voltage regulator. Under normal operating conditions, this will allow the generator control unit/voltage regulator to monitor and control the output voltage of the starter generator and in turn connect the output of the generator to the 28 VDC bus through the generator relay.

Positioning the generator switch to OFF opens the generator field circuit which removes control of the generator control unit/ voltage regulator from the generator and generator relay. The generator relay will open, removing the generator from the 28 DC bus.

In the event an over voltage condition is detected by the generator control unit/voltage regulator, an internal regulator trip relay circuit will be activated. Positioning the generator switch to RESET provides bus power, from the battery, to the generator control unit/voltage regulator which will reset the internal trip relay circuit. If the malfunction condition persists following the RESET, further attempts to reset should not be made.

GPU is disconnected prior to positioning the generator switch to GEN. Positioning the generator switch to GEN with the GPU connected may cause a reverse current situation and trip the generator off line.

1.32.E Start Switch

The start switch is located on the collective switch box. The switch contains four spring loaded poles that provide momentary contact to START position.

1.32.F Electrical System Indicators

Electrical system indicators include a DC ammeter, voltmeter, BATTERY RELAY light, START light, and GEN FAIL light.

1.32.G Fuel Pressure/ DC Ammeter

The fuel pressure/ammeter gauge is a dual display instrument. The ammeter indicates the load in amperes that is being supplied to the 28 VDC bus by the engine driven generator.

1.32.G.1 Ammeter – 400 Amps Max Scale

Ammeter indications will be continuous regardless of load. Refer to the FMS-E407-789-1 for generator load limitations.

To ensure limits are not exceeded, pilot can switch generator to OFF prior to generator exceedance being reached. Following a brief time, generator switch can be positioned to GEN (ON).

1.32.H Voltmeter

The voltmeter is included in a multifunction indicator mounted in the upper left area of the instrument panel. The indicator also displays Outside Air Temperature and Clock functions.

A button located on the center top of the instrument changes the top display between volts (e.g., 28 E) and OAT (Celsius and Fahrenheit). When power is applied to the instrument the display defaults to the voltmeter reading.

The voltmeter display receives its power from the 28 VDC bus through the OAT/V INSTR circuit breaker. The 28 VDC bus through this circuit breaker is also the source of the voltage value read and displayed by the voltmeter. When all electrical power is turned off, the voltmeter display disappears.

1.32.I Deleted

1.32.J Battery Relay Annunciator

The BATTERY RELAY light will be ON if the battery relay has remained in the closed (energized) position after the battery switch has been set to OFF.

If the battery relay remains energized after the battery switch has been set to OFF, battery power will remain on the 28 VDC bus. This will power the caution and warning panel to allow illumination of the BATTERY RELAY light, even if the generator is off.

1.32.K Start Annunciator

The START light will be illuminated when the starter relay is energized. The starter relay will be energized when the start switch is positioned to START. FADEC/ECU sends ARINC data to NP/NR that provides ground discrete to turn the START light on.

1.32.L Gen Fail Annunciator

The GEN FAIL light is controlled by generator relay. The light will be ON when the generator relay is de-energized and not connecting the generator output to the DC bus.

The generator relay will be energized by the generator control unit/voltage regulator when generator output climbs through a threshold of 24 ±2.4 VDC. Prior to the generator relay being energized, the GEN FAIL light will be ON. Once the generator relay is energized, the GEN FAIL light will be OFF.

1.33 Pitot Static System

The pitot-static system (Figure 1-22) utilizes a conventional impact air and ambient air pressure sensing system.

The pitot-static system consists of a heated pitot tube and right and left heated static ports and interconnecting tubing.

The pitot tube supplies impact air pressure to the airspeed indicator and PRCU (paragraph 1-30-D). The two static ports are connected together to equalize the static pressure, which is supplied to the airspeed indicator, altimeter, vertical speed indicator, and PRCU.

1.34 Basic Flight Instruments

The basic set of flight instruments includes an airspeed indicator, pressure altimeter, vertical speed indicator, and inclinometer.

1.34.A Airspeed Indicator

The airspeed indicator presents airspeed from 0 to 150 knots. The indicator is scaled in 20 knot increments from 0 to 20 knots and in 5 knot increments from 20 to 150 knots. A maximum speed red line is located at 140 knots and a maximum autorotation speed red/ white line is located at 100 knots.

1.34.B Altimeter

The pressure altimeter presents an altitude reading in feet above mean sea level (MSL) based on the relationship between the static air pressure and the barometric setting on the altimeter. The barometric setting may be adjusted to reflect the current barometric pressure corrected to sea level in inches of mercury or millibars.

1.34.C Vertical Speed Indicator (VSI)

The vertical speed indicator presents rate of climb or descent from 0 to 4000 feet per minute. A maximum rate of climb red line is located at 2000 FPM.

1.34.D Inclinometer

The inclinometer consists of a curved glass tube, ball, and damping fluid. The ball indicates the directional balance of the helicopter. If the helicopter is in a slip or a skid, the ball will move off center.

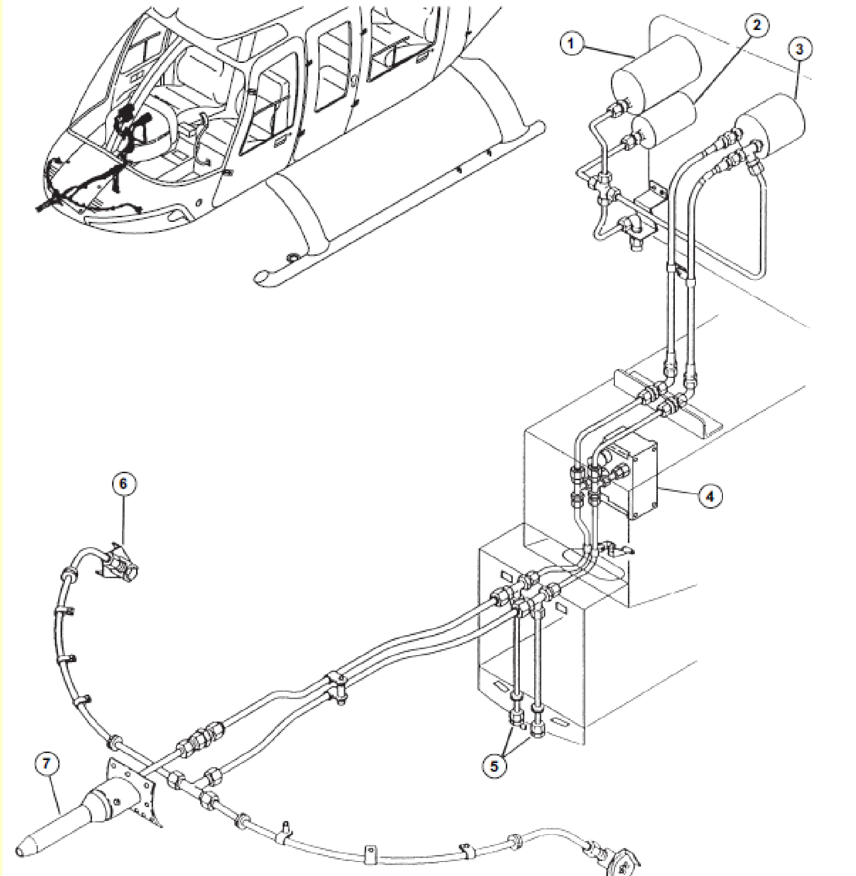


Figure 1-23 Pitot Static System

1.35 Navigation Systems and Instruments

The basic navigation equipment consists of a magnetic compass.

1.35.A Magnetic Compass

The magnetic compass is a standard, non-stabilized, magnetic type instrument mounted on a support which is attached to the right side of the forward crew cabin. The compass is used in conjunction with the compass correction card.

1.35.B Optional Navigation Equipment

Optional navigation equipment currently available through Bell Helicopter consists of a horizontal situation indicator (HSI), global positioning system (GPS), VOR indicator, automatic direction finder (ADF), and transponder. Basic operational information is provided for each installation. For expanded information, refer to the equipment manufacturer's operating manual.

1.35.B.1 Horizontal Situation Indicator (HSI)

The optional KI-525A horizontal situation indicator (HSI) provides a display of the gyro stabilized magnetic heading and helicopter position relative to the VOR and localizer and glide slope beams. Global positioning system (GPS) course information, if applicable, may also be displayed on the HSI. Switching between NAV or GPS is selected through a NAV/GPS switch.

Adjustments on the HSI include a heading select knob for setting the desired magnetic heading, and a course select knob for setting the desired course. The course deviation bar indicates displacement from the selected radial, track, or ILS localizer beam. The glideslope pointer indicates displacement from the ILS glideslope beam. An ambiguity pointer indicates whether the selected course will lead TO or FROM the station. The NAV fail flag warns the pilot of weak or unreliable VOR/ILS/GPS navigation signals and the HDG flag warns of gyrocompass failure.

The Bendix/King KCS 55A Gyromagnetic compass system is powered from the 28 VDC bus and provides magnetic heading information, which is displayed on the HSI. The system includes a KG-102A slaved directional gyro, KMT-112A magnetic azimuth transmitter, KA-51B slaving accessory, and a KI-525A HSI indicator.

The KA-51B slaving accessory control panel, located on the pedestal, provides two toggle switches and a slaving indicator. The SLAVE/ FREE switch engages the slave gyro mode when in SLAVE position and places the gyro in the free mode when in the FREE position. The CW/CCW switch is a momentary switch which provides clockwise and counterclockwise manual slewing when the system is in the free gyro mode.

A slaving meter on the control panel indicates the instantaneous error between the compass card presentation and the signal from the flux valve.

1.35.B.2 Global Positioning System (GPS)

GPS is a Department of Defense (DOD) operated global coverage, satellite-based navigation system.

The optional KLN 89B GPS is a long-range, GPS based RNAV System. It consists of a panel-mounted receiver/display unit and one antenna. The system is designed to supply the Pilot with navigation guidance and position information in three-dimensions: latitude, longitude, and altitude.

The NAV/GPS push button on the instrument panel (if installed) will configure the HSI to depict VOR/LOCALIZER/GLIDESLOPE information when selected to NAV and GPS information when selected to GPS.

Additionally, pressing the caution panel test button will turn on the NAV/GPS switch lights.

Refer to appropriate FMS and the Bendix/King (Allied Signal) KLN 89B Pilots Guide for additional information.

1.35.B.3 VOR Indicator

An optional KI-208 course deviation indicator may be installed as a primary

VOR/LOCALIZER indicator.

An omni bearing selector (OBS) and TO/ FROM ambiguity pointer are provided. The NAV fail flag warns of weak or unreliable VOR, localizer signals, or equipment malfunction.

1.35.B.4 Automatic Direction Finder (ADF)

The optional automatic direction finder set includes a KR 87 ADF receiver, KI 227 ADF indicator, and a KA 44B loop/sense antenna. The receiver includes active/standby frequency selection and a flight or elapsed timer.

The right-hand side of the KR 87 receiver will display either the standby frequency or the flight or elapsed timer. If it is in one of the timer modes, press the FRQ button once to display the standby frequency. Press the FRQ button again to exchange the active and standby frequencies.

Press the FLT/ET button to return to one of the timer modes. The unit will enter whichever timer mode was displayed previous to pushing the FRQ button. Therefore, if it enters the ET (elapsed timer) mode, press the FLT/ET button once more to enter the FLT (flight timer) mode. The flight timer should reset to zero if the unit is turned off and turned back on.

The elapsed timer (ET) has two modes: count-up and count-down. If the display is in the FLT mode, press the FLT/ET button once to display the elapsed timer. When power is applied, the ET is in the count-up m ode starting at 0. When in the count-up mode, the timer may be reset to 0 by pressing the RESET button.

To enter the count-down mode, hold the RESET button in for approximately 2 seconds until the ET message begins to flash. The display may now be set to any time up to 59 minutes and 59 seconds. The timer will remain in this ET set mode (whenever ET message is flashing) for 15 seconds after a number is preset, or until the RESET, FLT/ET, or FRQ button is pressed. The preset number will remain unchanged, until the RESET button is pressed, at which time it will begin to count down. When the time reaches 0, it will begin counting up from 0 and the display will flash for 15 seconds (regardless of the current display). While the elapsed timer is counting down, pressing the RESET button will have no effect unless it is held for 2 seconds, putting the timer into the ET set mode.

Pressing the FLT/ET button will exchange the two timers in the display, or will cause the last timer that was displayed to reappear if the standby frequency is being displayed. Pressing the FRQ button will cause the standby frequency to reappear, and subsequent actuation will cause the active and standby frequencies to be exchanged.

1.35.B.5 ATC Transponder

The optional KT-70 or KT-76A Transponder is a radio transmitter and receiver operating at 1090 MHz in transmit mode and 1030 MHz in receive mode. The equipment is designed to fulfill the role of airborne beacon under the requirements of the Air Traffic Control Radar Beacon System (ATCRBS). Range and azimuth are determined by the transponder pulsed return in response to interrogation from the ground radar site. An identity code number, selected at the front panel, is transmitted at a Mode A reply. For mode C altitude reporting capability, the transponder must be used in conjunction with a reporting (encoding) altimeter and operated in ALT mode. After pressing the IDENT button when interrogated, the transponder will transmit a special pulse causing the associated PIP to bloom on the ATC display. Either transponder can reply on any of 4096 preselected codes.

1.35.B.6 VHF NAV/COMM System

The optional VHF NAV/COMM system can include a KX-155 NAV/COMM, or KX-1 65 NAV/COMM as COMM 1 and KY196A COMM 2.

The KX-155 and KX-165 are VHF NAV/COMM transceivers which operate within the frequency range of 118.00 MHz to 136.975 MHz, in 25 kHz increments (760 channels) for COMM, and 108.00 MHz to 117.95 MHz, in 50kHz increments (200 channels), for NAV.

Both the KX-155 and KX-165 NAV/COMM have two displays: the left-hand display for COMM frequencies and the right-hand display for NAV frequencies. The COMM display presents a USE and STANDBY frequency with a T appearing between the frequency numbers to indicate operation in the transmit mode. The NAV display presents a USE and STANDBY frequency. For both COMM and NAV displays, the desired frequency is entered into the STANDBY window and transferred to the USE window by depressing the transfer button. Both the COMM and NAV, and USE and STANDBY frequencies are stored in a NVM on power down, and will be displayed when the unit is turned on. If an invalid frequency is detected in the memory when power is applied, the COMM USE and STANDBY windows will display 120.000 and the NAV USE and STANDBY windows will display 110.00

In addition, when the smaller NAV frequency selector knob is pulled on the KX-165, the radial of the active VOR is displayed in the STANDBY window.

The KY196A is a VHF COMM transceiver which operates within the frequency range of 118.000 MHz to 136.975 MHz in 25 kHz increments (720 channels). The KY 196A display presents a USE and STANDBY frequency with a T appearing between the frequency numbers to indicate operation in the transmit mode. The desired frequency is entered into the STANDBY window and transferred to the use window by depressing the transfer button. The COMM USE and STANDBY frequencies are stored in a NVM on power down, and will be displayed when the unit is turned on.

1.36 Three or Five Place Intercommunication System

Optional intercommunication and audio distribution is accomplished by one KMA 24H-71 audio control panel.

The KMA 24H-71 has separate speaker and headphone isolation amplifiers which are powered by separate circuit breakers on the avionic bus, providing a high degree of audio integrity. The system is equipped with primary and secondary headphone amplifiers serving the pilot and the crew/intercom stations respectively. This provides for the pilot and/or copilot to isolate from the intercom system. Headphone outputs contain the audio selected by the

PHONE pushbuttons and the MIC rotary selector switch.

The system requires headsets or helmets with a 300 ohm impedance rating. Military type helmets with an 8 ohm impedance rating are not compatible and may cause damage to the audio control panel.

The KMA 24H-71 audio panel MIC rotary selector switch has seven positions which are normally connected as follows:

POSITION	FUNCTION
EMG	Emergency
1	VHF No.1
2	VHF No.2
3	Not used
4	Not used
5	Not used
PA	Aft Cabin Speaker

The KMA 24H-71 audio panel mixing push button switches are normally connected as follows:

SWITCH	FUNCTION
COMM 1	VHF No.1
COMM 2	VHF No.2
COMM 3	Not Used
COMM 4	Not used
COMM 5	Not used
NAV 1	VOR/ILS
NAV 2	Not Used
DME	Not Used
MKR	Not Used
ADF	ADF
SPKR AUTO	Audio from SPKR PULL OUT Switch

The ICS mode rotary selector switch, located on the center console, has three mode positions: NORMAL, ISOLATE, and PRIVATE. Operation in the three modes is as follows.

1.36.A Normal Mode

With the ICS mode rotary selector switch positioned to NORMAL, and the KMA 24H-71 audio panel MIC rotary select or switch positioned to COMM 1 or COMM2 (if applicable), all headphones will receive audio from all ICS inputs and selected audio control panel mixing push button selections.

To operate in the NORMAL (keyed ICS) mode, the VOX keying knob on the KMA 24H-71 audio panel is to be turned fully counter clockwise. This will allow audio to all headsets when the pilot or copilot cyclic switch is keyed to the first position, the copilot foot switch is keyed, or any of the aft ICS drop chord switches are keyed.

To operate in the NORMAL (Hot MIC) mode, the VOX keying knob on the audio panel is to be turned fully clockwise. This will allow audio to all headsets by voice activation from any position.

1.36.B Isolate Mode

With the ICS mode rotary selector switch positioned to ISOLATE, and the KMA 24H-71 audio panel MIC rotary select or switch positioned to COMM 1 or COMM 2 (if applicable), the pilot will be isolated from the intercom. Communication between the copilot and aft ICS stations will be available and audio from the pilot shall not be heard in the copilot, aft cabin speaker, or aft ICS headsets. Audio control panel mixing push button selections will only be heard through the pilot headset.

To operate in the ISOLATE (keyed ICS) mode, the VOX keying knob on the KMA 24H-71 audio panel is to be turned fully counter clockwise. This will allow audio between the copilot and aft ICS positions when the copilot cyclic switch is keyed to the first position, the copilot foot switch is keyed, or any of the aft ICS drop chord switches are keyed.

To operate in the ISOLATE (Hot MIC) mode, the VOX keying knob on the audio panel is to be turned fully clockwise. This will allow audio between the copilot and aft ICS positions by voice activation.

1.36.C Private Mode

With the ICS mode rotary selector switch positioned to PRIVATE, and the KMA 24H-71 audio panel MIC rotary selector switch positioned to COMM 1 or COMM 2 (if applicable), the pilot and copilot will be isolated from the aft ICS stations.

Hot MIC (voice activated) audio is automatically provided to all positions in the PRIVATE mode regardless of the VOX keying knob position. In addition, audio from the audio control panel mixing push button selections will only be heard by the pilot and copilot.

1.36.D Aft Cabin Mode

Aft cabin speaker audio may be selected by positioning the KMA 24H-71 audio panel MIC rotary selector switch to PA. Keying the pilot or copilot cyclic stick switch to the second position (radio) will allow audio to be heard in the aft cabin speaker. This will occur with the ICS mode rotary selector switch positioned to NORMAL, ISOLATE, or PRIVATE.

1.37 Emergency Communication System

In the event of an intercommunication system failure, emergency communication is available by positioning the KMA 24H-71 audio panel MIC rotary selector switch to EMG. With VHF COMM 1 operating and selected to the required frequency, keying the pilots cyclic stick switch to the second position (radio) will allow two-way communication. Audio will only be available in the pilot’s headset.

Additionally, the emergency communication system can be activated with the ICS mode rotary selector switch positioned to NORMAL, ISOLATE, or PRIVATE.

1.38 Avionics Master Switch

The AVIONICS MASTER switch allows activation and deactivation of all avionics components simultaneously. Circuit breakers that are connected to the avionics master switch are identified by triangles next to the circuit breakers on the overhead console. The main 28 VDC bus powers COMM 1 (VHF 1 NAV/COMM) and the audio panel (ICS PHONE).

1.39 Miscellaneous Instruments

Miscellaneous instruments include a combined outside air temperature, clock, and voltmeter indicator, and an hourmeter.

1.39.A Outside Air Temperature Indicator

The outside air temperature is included in a multifunction indicator mounted in the upper left area of the instrument panel. The indicator also displays voltmeter and clock functions.

A button located on the center top of the instrument changes the top display between OAT (Celsius and Fahrenheit) and volts (e.g. 28E). When power is applied to the instrument the display defaults to the voltmeter reading. Pushing the button will change the display to temperature.

The temperature is taken by a probe mounted outside the helicopter in the lower nose section.

When all electrical power is turned off, the outside air temperature display disappears.

1.39.B Hourmeter

An hourmeter is located on the aft wall of the battery compartment. It can only be viewed from the inside of the battery compartment. The hourmeter is a digital instrument that registers cumulative time, in hours and tenths. The hourmeter is powered by the hourmeter circuit breaker located in the battery compartment. For the hourmeter to record time, the hourmeter circuit breaker must be in, the engine NG must be greater than 55%, and the helicopter weight must be off the landing gear (the helicopter must be in-flight).

1.39.C Ventilation and Defog System

A vent and defog system is installed at each crew station. Each side consists of a plenum with a vent door, electric blower, windshield defog nozzle, and a control cable.

Control cables, with knobs, are installed below either side of the instrument panel. Pulling the cable opens the exterior vent door to allow outside ram air to enter the cabin through an opening at the bottom of the plenum. The control cables lock in any position and are released by pressing the center button on each knob.

An electrically-driven axial flow blower in each system provides airflow for ventilation and defogging when the helicopter is on the ground or hovering. The blower intake takes air from the cabin and blows it through the windshield defog nozzle onto the windshield. Both blowers are controlled by one DEFOG switch located in the overhead console.

1.39.D Lighting System

The lighting systems include interior and exterior lighting. The interior lighting system includes a cockpit utility light, instrument panel and associated lighting, and aft cabin lighting. The exterior lighting system includes landing, position, and anticollision lighting.

1.39.E Cockpit Utility Light

A removable utility light is secured in a bracket on the forward side of the control tube tunnel between the cockpit seat backs. A long, spiral wound cord permits use anywhere in the cockpit.

The light will provide a blue or white light depending on the setting of the selection switch. It can be used as a spot or flood light and the intensity of the light is controlled with the BRT/DIM control knob. Power to the cockpit utility light is provided through a 5-amp CKPT LIGHTS circuit breaker.

1.39.F Instrument Panel and Associated Lighting

The instrument lights are powered by 28 or 5VDC. An INSTR LIGHTS circuit breaker provides 28 VDC directly to certain instruments and also to a 5 volt power supply. Some instruments (primarily propulsion instruments) are lighted by the 5 volt power supply. Other instruments (primarily flight instruments) are powered by 28 VDC. All instrument lights are controlled by the INSTR LT rheostat located in the overhead console. The caution panel is supplied with 28V in bright and 15V in dim.

The caution/warning panel dimming is restricted to one position 15V dimming. To dim the caution panel, the INSTR LT rheostat must be positioned between dim and bright. Momentarily positioning the CAUT LT dim switch to DIM will dim the caution panel lights to a fixed dim mode.

Caution/warning panel segments FLOAT TEST, ENG OIL PRESS, ENGINE OVSPD, ENGINE OUT, and RPM are not dimmable. To test all of the caution panel segment lamps, press the CAUTION LT TEST switch. This will also test the lamps for the FADEC mode switch and NAV/GPS, and QUIET MODE switch lamps (if installed). The PEDAL STOP switch must be pressed to test its internal lamps.

1.39.G Aft Cabin Lighting

The aft cabin lighting system consists of two individual reading lights. Power is provided to the lights from a 5-amp CKPT LIGHTS circuit breaker and controlled through the CABIN/ PASS LT switch and two individual reading light switches.

With the CABIN/PASS LT switch positioned to OFF, all cabin lights will be OFF regardless of the position of the two reading light switches. With the CABIN/PASS LT switch positioned to CABIN LT, all cabin lights will be ON regardless of the position of the two reading light switches. Positioning the CABIN/PASS LT switch to PASS LT will allow control of the reading lights through the reading light switches.

1.39.H Landing Lights

The landing lights consist of one forward and one downward facing lamp. Both lamps are exposed for improved cooling.
Power is provided to the landing lights through separate relays which are controlled by the LDG LIGHTS switch located on the collective switch box. A 2-amp LDG LT CONT circuit breaker is used to power the coils of each control relay and a 25-amp LDG LT POWER circuit breaker is used to power the landing lights through the control relays.

Positioning the LDG LIGHTS switch to FWD will turn on the forward landing light and positioning the switch to both will turn on both the forward and downward landing lights.

1.39.I Position Lights

The position lights consist of a green light located on the horizontal stabilizer right vertical fin, a red light on the horizontal stabilizer left vertical fin, and a white light located on the tail. Position lights are also located on the lower cabin with a green light on the right side, and a red light on the left side.

Power is provided to the position lights through a 5-amp POS LT circuit breaker switch located on the overhead panel.

1.39.J Anti-collision Light

The anti-collision light consists of a single red strobe light mounted on top of the vertical fin. Power is provided to the strobe from a separate power supply, which is controlled by a 5-amp ANTI COLL LT circuit breaker switch located on the overhead panel.

1.40 Emergency Equipment

Emergency equipment includes a portable fire extinguisher, a first aid kit, and optional Pointer 4000 ELT kit.

1.41 Portable Fire Extinguisher

A portable fire extinguisher is mounted between the cockpit seat backs.

1.42 First Aid Kit

The first aid kit is supplied as loose equipment.

1.43 Pointer 4000 ELT

The Pointer 4000 ELT installation includes a 4000-10 transmitter installed on the left side of the pedestal, a remote switch on the right side of glare shield, and an external antenna mounted on the forward upper cowl.

The Pointer ELT is a self-contained emergency transmitter capable of manual or automatic operation. It is designed to withstand forced landing and crash environment conditions. Automatic activation is accomplished by a deceleration sensing inertia switch. The inertia switch is designed to activate when the unit senses longitudinal inertia forces, as required in TSO-C91A.

To configure the ELT to automatically activate with the remote switch installed, the Master switch on the ELT must be set to AUTO and the remote switch must be set to AUTO. This will allow the ELT to activate when the inertia switch senses predetermined deceleration level.

To override the inertia switch and turn on the ELT manually, the remote switch must be positioned to ON. This procedure may be used during unit testing or if an emergency situation is imminent and pilot wishes to activate ELT prior to emergency.

The RESET position of the remote switch is used to deactivate and re-arm the ELT transmitter to AUTO mode after automatic activation by the inertia switch. Helicopter power is required for remote reset. In case of inadvertent activation of ELT transmitter with helicopter power OFF, turn helicopter power ON, position remote switch to RESET, and then back to AUTO.

After a forced landing or helicopter accident, if helicopter receiver is operable, listen on 121.5 MHz for ELT transmissions. The range of ELT varies according to weather and topography. In general, the swept tone signal can be heard up to 30 miles by a search helicopter at 10,000 feet. It is recommended to stay close to the helicopter to permit easier spotting by airborne searches.

It may also be desirable to use ELT transmitter in the portable mode due to a broken or disabled whip antenna, severed antenna coax cable, danger of fire explosion, temperatures extremes in helicopter, poor transmitting location, or water ditching with forced evacuation.

To remove transmitter from helicopter:

1. Bend switch guard away from unit Master Switch and place switch in OFF position.
2. Disconnect remote antenna coax cable.
3. Disconnect remote switch cable.
4. Remove telescopic antenna from stowage clips. Unlatch ELT transmitter hold down strap and remove unit from bracket.
5. Insert telescopic antenna into ANT receptacle. Extend antenna fully.
6. Turn ELT transmitter Master switch to ON position. DO NOT USE AUTO POSITION

Consider factors such as terrain, temperature, and precipitation when choosing a location for the ELT transmitter to radiate from. The best transmission may be obtained by keeping the antenna vertical and setting transmitter upright on a metallic surface. If terrain prohibits good transmission (such as deep valley or canyon), place the transmitter on high ground or hold in hand in high place.

As cold temperatures have a direct effect on ELT transmitter battery life, the life of the battery pack can be extended by placing the ELT transmitter inside a jacket or coat to keep the battery warm. Let antenna extend outside jacket. Keep all moisture and ice away from the antenna connection and the remote connector pins.

Ensure ELT transmitter is turned ON continuously (day and night), until rescue team appears.

For additional information, refer to the Pointer Aircraft Emergency Locator Transmitter Operation and Installation booklet.

Section 2

Handling and Servicing

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Section 2

Handling and Servicing

2.1 Ground Handling

Ground handling of the helicopter consists of towing, parking, securing, and mooring. Model 205 or 206 ground handling wheels are used for towing. Refer to the ICA-E407-789, Chapter 9 for more detailed ground handling information.

CAUTION

DO NOT TOW THE HELICOPTER IF THE GROSS WEIGHT IS MORE THAN 5000 POUNDS (2270 KG) FOR MODEL 205 GROUND HANDLING WHEELS, OR 4450 POUNDS (2020 KG) FOR MODEL 206 GROUND HANDLING WHEELS.

2.2 Covers and Tie Downs

Protective covers and tie-downs are furnished as loose equipment and are used for parking and mooring of helicopter (Figure 2-1). Additional equipment such as ropes, cables, clevises, ramp tie-downs, or dead man tie-downs will be required during mooring.

2.2.A Cover – Engine Inlet

Engine inlet plug assemblies are red and flame resistant, and each cover is attached with a red streamer stenciled in white letters, REMOVE BEFORE FLIGHT. To install the engine inlet plug, make sure that the side marked TOP is facing upwards. Push the engine inlet plug into the engine air inlet.

2.2.B Cover – Pitot Tube

WARNING

PITOT TUBE CAN BE HOT.

Pitot tube cover assembly is red, flame resistant, and attached with a red streamer stenciled in white letters, REMOVE BEFORE FLIGHT. To install pitot tube cover, push it over the pitot tube. Attach the cord.

2.2.C Cover – Engine Exhaust

Engine exhaust cover is red and flame resistant, and includes a red streamer stenciled in white letters, REMOVE BEFORE FLIGHT. A ¼ inch diameter elastic tie-cord is attached to cover for securing to engine exhaust. To install the engine exhaust cover, push it over the exhaust tailpipe. Attach the tie-cord.

2.2.D Cover – Oil Cooler Blower Inlet Duct

Oil cooler blower inlet duct plug assemblies are red and flame resistant, and each cover is attached with a red streamer stenciled in white letters, REMOVE BEFORE FLIGHT. To install the inlet plug, push it into the oil cooler blower inlet duct.

2.2.E Tie Down – Main Rotor

For each main rotor blade, there is a main rotor tie-down assembly. Each tie-down assembly has a sock assembly and a line assembly. Use these assemblies to attach the blades to the landing gear cross tubes. The sock assembly is red and a red steamer attached to it stenciled with white letters: REMOVE BEFORE FLIGHT. The line assembly is made of 0.19 inch (4.83 mm) diameter nylon and has a ring and an attached flag. The flag is stenciled with the letters FWD BLADES or AFT BLADES.

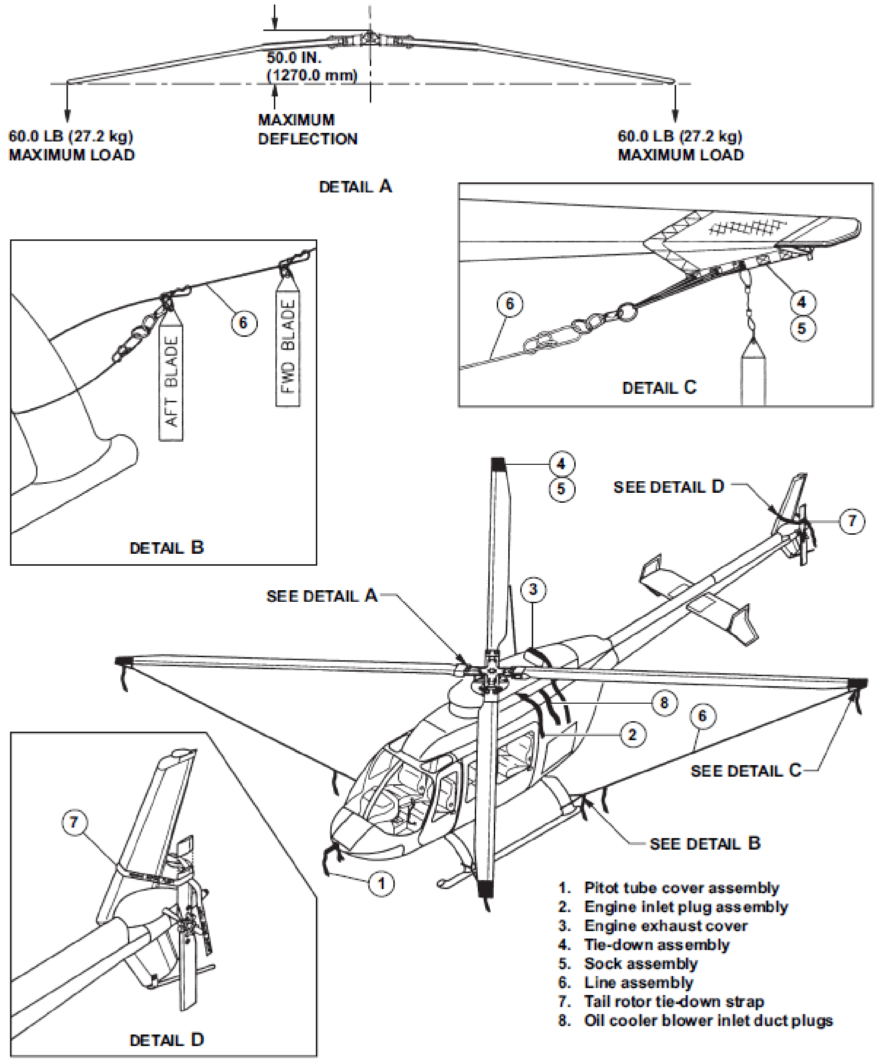

Figure 2-1. Covers and Tie-Downs

Install the main rotor tie-down assemblies as follows:

CAUTION

DO NOT CAUSE THE MAIN ROTOR BLADES TO BEND MORE THAN THE LIMITS SHOWN IN FIGURE 2-1 DETAIL A.

NOTE

At the same time that you align the main rotor blades, align the tail rotor blades with the vertical fin. This will make it possible to install the tail rotor tie-down.

1. Turn the main rotor blades until there are two blades aft of the fuselage station of the main rotor hub. When you look down at the helicopter, the four blades make an X over the vertical center line of the fuselage.
2. Install the two FWD BLADES sock assemblies on the ends of the main rotor blades that are forward of the fuselage station of the main rotor hub.
3. Put a line assembly around each outboard end of the forward crosstube of the landing gear.

NOTE

Rings are pre-set to apply the necessary tension to the forward and aft main rotor blades.

4. Attach the snaps of the two line assemblies to the rings of the two FWD BLADES sock assemblies.
5. Install the two AFT BLADES sock assemblies on the ends of the two aft main rotor blades.
6. Put a line assembly around each outboard end of the crosstube of the

landing gear.

7. Attach the snaps of the two line assemblies to the rings of the two AFT BLADES sock assemblies.

2.2.F Tie-Down – Tail Rotor

The tail rotor tie-down strap is made of 0.025 x 1.0 x 92.0 inches (0.635 x 25 x 2340 mm) nylon webbing. It is red and stenciled with white letters REMOVE BEFORE FLIGHT.

CAUTION

DO NOT TIE DOWN TAIL ROTOR TO EXTENT THAT TAIL ROTOR BLADE FLEXES. THE APPLIED FORCE OF THE TIE-DOWN SHOULD PROVIDE A LIGHT CONTACT BETWEEN THE TAIL ROTOR YOKE AND THE FLAPPING STOP.

1. Turn the main rotor blades until there are two blades aft of the fuselage station of the main rotor hub. When you look down at the helicopter, the four blades should make an X over the vertical center line of the fuselage. Align the tail rotor with the vertical fin.
2. Leaving enough strap material to extend around the vertical fin, wrap the tail rotor tie-down strap around the upper tail rotor blade once (Figure 2-1, Detail D).
3. Position the loose ends of the strap around the upper half of the vertical fin and tie the two ends together

2.2.G Parking – Normal and Turbulent Conditions

When winds are forecast to be light or up to 50 knots, park helicopter pointed in direction from which you expect the highest winds.

Moor helicopter as follows:

1. Hover, taxi, or tow helicopter to the specified parking area.

2. Remove ground handling gear (if installed).
3. Attach the main and tail rotor blade tie-downs.
4. Install the engine air inlet plugs, oil cooler blower inlet duct plugs, pitot tube, and the engine exhaust covers.
5. Tighten the friction locks on the flight controls.
6. Make sure that all switches are in the OFF position.
7. Disconnect the battery.
8. Close and safety all of the doors, windows, cowlings, and access panels.
9. If helicopter is parked outside in a heavy dew environment, purge lubricate all of the control bearings that are open to the air. Do this once every 7 days. Make sure no voids exist that could trap moisture.

2.2.H Mooring (Winds Above 50 Knots)

When winds above 50 knots are forecast, park helicopter pointed in direction from which you expect the highest winds.

Moor helicopter as follows:

CAUTION

WHEN WINDS ABOVE 75 KNOTS ARE FORECAST, PUT THE HELICOPTER IN A HANGAR OR MOVE IT TO AN AREA WHERE IT WILL NOT BE AFFECTED BY THE WEATHER. FLYING OBJECTS DURING HIGH WINDS CAN CAUSE DAMAGE TO THE HELICOPTER.

NOTE

If the correct ramp tie-downs are not available, park the helicopter on an unpaved area. Use the dead man tie-downs. Point the helicopter into the wind and remove the ground handling wheels.

1. Attach helicopter to the ramp tie-downs.

NOTE

Use a mooring clevis at each of the three jack fittings. This will let you use a rope with a larger diameter.

2. Attach the cable, rope, or the manufactured tie-downs to the helicopter jack fittings.

CAUTION

DO NOT CAUSE THE MAIN ROTOR BLADES TO BEND MORE THAN 50.0 INCHES (1270 MM), AS SHOWN IN FIGURE 2-1.

3. Attach the main and tail rotor tie-downs.
4. If time and storage space are available, remove the main rotor blades and put them in a safe building.

NOTE

Put all of the red streamers inside an access door so that they will not flap in the wind.

5. Install the engine air inlet plugs, oil cooler blower inlet duct plugs, pitot tube cover, and engine exhaust cover.
6. Tighten friction locks on the flight controls.

CAUTION

MAKE SURE THAT ALL OF THE SWITCHES ARE IN THE OFF POSITION, AND THAT ALL OF THE CIRCUIT BREAKERS ARE OPEN.

7. Disconnect the battery.
8. Close and safety all of the doors, windows and access panels.
9. Refuel the helicopter to its maximum capacity.

CAUTION

SAFETY OR REMOVE ALL OF THE EQUIPMENT AND OBJECTS IN THE AREA. OTHERWISE, THE WIND CAN BLOW THE OBJECTS AGAINST THE HELICOPTER AND CAUSE DAMAGE.

10. Safety or remove all of the equipment and objects in the area.
11. When the winds stop, examine the helicopter for damage.

2.3 Fuels

Operation of the HTS-900-2-1D engine on other than jet fuel is not recommended. However, some fuels may be considered acceptable for limited use on an emergency basis with certain restrictions and precautions. The HTS900-2-1D engine is not approved for use with any highly volatile emergency fuels such as aviation gasoline.

Fuels conforming to the following commercial and military standards are approved.

Specification	OAT Range
ASTM D-1655, Jet A or A-1	Above -32°C (-25°F)
ASTM D-6615, Jet B	Any OAT
MIL-DTL-5624, Grade JP-4 (NATO F-40)	Any OAT
MIL-DTL-5624, Grade JP-5 (NATO F-44)	Above -32°C (-25°F)
MIL-DTL-83133, Grade JP-8 (NATO F-34)	Above -32°C (-25°F)

Refer to FMS-E407-789-1 for fuel limitations and use of anti-icing additives. Fuel listings (Table 2-9) are provided for convenience of operator. It shall be the responsibility of the operator and the fuel supplier to ensure fuel used in helicopter conforms to one of approved specifications.

Refer to Honeywell Light Maintenance Manual for HTS900-2-1D for alternate or emergency fuels.

2.3.A Fuel System Servicing

Total capacity	130.5 U.S. gallons (493.9 L)
Usable fuel	127.8 U.S. gallons (483.7 L)
Unusable fuel	2.7 U.S. gallons (10.2 L)
Undrainable fuel (included in unusable fuel)	0.7 U.S gallons (2.6L)

Fuel system contains two interconnected cells that are serviced through a single fuel port located on right side of helicopter. A grounding jack is provided near fueling port. An electric sump drain is located in both the forward and aft tanks. They are activated by buttons located on the right aft lower side of fuselage. Battery switch must be ON (or external power applied) and fuel valve switch must be OFF to activate sump drains.

2.3.B Fuel Operational Additives

WARNING

EACH OF THE ADDITIVES DESCRIBED HEREIN IS AN IRRITANT AND A HEALTH HAZARD. REFER TO THE MATERIAL SAFETY DATA SHEET FROM EACH MANUFACTURER FOR SPECIFIC HEALTH, SAFETY AND STORAGE INSTRUCTIONS.

CAUTION

TO BE EFFECTIVE, ALL ADDITIVES MUST BE USED IN STRICT ACCORDANCE WITH MANUFACTURER'S INSTRUCTIONS.

Where permitted, the fuels defined in Table 2-9 may be blended with the additives described in Table 2-1 through 2-7, either singly or in combination at the approved concentration.

2.3.B.1 Anti-Static Additives

Static-dissipater additives are blended into fuels to prevent accidental discharge of static electricity during transfer of the fuel from the storage tanks or transporters to the aircraft. Most military fuels and civil aviation fuels from some major manufacturers have static-dissipater additives automatically blended into the fuel. However, some commercial fuel specifications do not require static-dissipater additives. If the helicopter requires such additives for the fuels, the additives must be blended in the amount specified by the manufacturer, but may not exceed the electrical conductivity limits allowed by the appropriate specifications. The static-dissipater additive brands and approved concentrations are defined in Table 2-1.

Table 2-1. Honeywell Approved Anti-Static Additives

Brand[a]	Comments	Maximum Concentration
Stadis 450[b][c][d]	Total accumulated at manufacture Cumulative total when redoping.	3 mg/L (or ppm) 5 mg/L (or ppm)
Sigbol (Russian) TU-38-101741-78	Total accumulated percent by weight.	0.0005%

[a] If an anti-static additive is used, electrical conductivity must conform to the following values:

1. EMS53112 Jet A-1: 50 to 450 pS/m JP-8: 200 to 600 pS/m
2. EMS53111 Jet A: 50 to 450 pS/m
3. EMS53113 Jet B: 50 to 450 pS/m JP-4: 200 to 600 pS/m

[b]Automatically blended into JP-8 and JP-4.

[c]Although some major manufacturers of civil aviation fuels automatically blend this additive, it is not required by the ASTM D 1655 specification for Jet A or Jet A-1.

[d]In situations in which redoping with static-dissipater additive is required to maintain conductivity and the original material is not available or not known, further addition of Stadis 450 must not exceed 2 mg/L.

2.3.B.2 Fuel Anti-Icing Inhibitors

CAUTION

ADD ANTI-ICE ADDITIVES TO FUELS AT OR BELOW AMBIENT TEMPERATURE OF 32°F (0°C).

Tables Table 2-2 through 2-4 list the approved fuel system icing inhibitor (FSII) additives for the HTS900-2-1D engine.

Table 2-2. Honeywell Approved EGME FSII Additives

EGME-Based FSII	Concentration, % by Volume
ASTM D 4171, Type 1	0.10 to 0.15
MIL-DTL-27686	0.10 to 0.15
AL-31 per DERD 2451	0.10 to 0.15
NATO Code S-748	0.10 to 0.15
Methyl Cellosolve	0.10 to 0.15

Table 2-3. Approved DiEGME FSII Additives

DiEGME-Based FSII	Concentration, % by Volume
ASTM D 4171, Type III	0.10 to 0.15
MIL-DTL-85470*	0.10 to 0.15
AL-41 per DEF STAN 68-252 (DERD 2451)	0.10 to 0.15
NATO Code S-1745	0.10 to 0.15
Prist (PPG) High-Flash Anti-Ice**	0.10 to 0.15

*Automatically blended into military JP-4, JP-5, and JP-8 fuels.
**Manufactured by Houston Chemical Co., PPG Industries under the name of Prist.

Table 2-4. Honeywell Approved Russian FSII Additives

Brand	Blend With	Comments	Concentration, % by Volume
I per GOST 8313-88 ethylene glycol monoethyl ether (Ethyl Cellosolve, Type A)	GOST 10227	Total accumulated percent by volume	0.10 to 0.20
TGF per GOST 17477-86 tetrahydrofurfuryl alcohol	GOST 10227	Total accumulated percent by volume	0.10 to 0.20
I-M per GOST TU-6-10-1458-79 (50/50 mixture by weight of I and methanol)	GOST 10227	Total accumulated percent by volume	0.10 to 0.20
TGF-M (GOST TU-6-10-	GOST 10227	Total accumulated	0.10 to 0.20

1458-79) (50/50 mixture by weight of TGF and methanol)	percent by volume

2.3.B.3 Fungicide/Biocide Inhibitors

CAUTION

DO NOT ADD MICROBIOCIDES TO JP-4, JP-5, OR JP-8 FUELS, OR TO NATO CODE F-34, F-40, OR F-44 FUELS OR TO ANY FUELS WHERE FUEL SYSTEM ICING INHIBITORS HAVE BEEN ADDED.

Microbiocides prevent contamination from biological growth in the helicopter fuel system under certain adverse environmental conditions. They are a complement to, and not a substitute for fuel system cleanliness.

Follow manufacturer's recommendations on mixing. Failure to properly mix microbiocides can result in engine fuel system and/or hot section damage. During microbiocides use, monitor fuel flow low-pressure warning signals and/or fuel pressure differential signals.

Continuous use of Biobor JF is not permitted due to the potential for hot section corrosion, because elemental Boron makes up 7.4 percent of Biopor JF. Continuous use of Kathon FP1.5 is not recommended due to the potential to develop resistant microorganisms.

Table 2-5 lists the approved fungicide and biocide inhibitors for the HTS900-2-1D engine.

Table 2-5. Honeywell Approved Fungicide/Biocide Inhibitors

Brand	Comments	Maximum Concentration
Biobor JF	Intermittent (kill) treatment	Not to exceed 270 ppm (20 ppm maximum of elemental boron)
KATHON FP 1.5	Intermittent (kill) treatment	100 ppmw
Methyl Collosolve (EGME) or Prist (DiEGME)	Continues use.	0.15% by volume

2.3.B.4 Fuel Troubleshooting Additives

WARNING

YELLOW DYE (AUTOMATE YELLOW 662) IS FLAMMABLE.

Table 2-6. Honeywell Approved Fuel Troubleshooting Additives

Brand	Blend With	Comments	Maximum Concentration
Automate Yellow 662	All	Used for verification of rotorcraft fuel leaks	1.6 ounces per 100 gallons of fuel

2.3.B.5 Anti-Corrosion Additives

Table 2-7 lists the approved anti-corrosion additives per ASTM D 1655 or MIL-PRF-25017 listed in QPL-25017-22 that may be mixed with any fuel for the HTS900-2-1D engine. Military jet fuels JP-4, JP-5, and JP-8 already contain an anti-corrosion additive.

Table 2-7. Honeywell Approved Anti-Corrosion Additives

Brand	Minimum Dosage*	Maximum Dosage
Apollo PRI-19	18 g/m^3	22.5 g/m^3
Innospec DCI-4A	9 g/m^3	22.5 g/m^3
Afton Hitec 580	15 g/m^3	22.5 g/m^3
Nalco 5403	12 g/m^3	22.5 g/m^3

*The minimum dosage ensures that sufficient additive is available for lubricity aid. Note: 1 g/m^3 = 1 mg/L

2.4 Oils

Approved oils and vendors are listed in this section for convenience of operator.

An appropriate entry shall be made in helicopter logbook when oil has been added to engine, transmission, or tail rotor gearbox. Entry shall show specification and brand name of oil used to prevent inadvertent mixing of oils.

CAUTION

DO NOT MIX OILS OF DIFFERENT SPECIFICATIONS. IF OILS BECOME MIXED, SYSTEM SHALL BE DRAINED, FLUSHED, AND REFILLED WITH PROPER SPECIFICATION OIL.

2.4.A Engine Oils

Certain oils conforming to the following specifications are approved for use in engine:

Table 2-8. Honeywell Approved Engine Oils

SPECIFICATION	Oil Brand and Trade Name
MIL-PRF-23699 (OAT above -40°C/-40°F)	BP Turbo Oil 2380 (Exxon Turbo Oil 2380)
	Mobile Jet Oil II
	Mobile Jet Oil 254

Engine oils (Table 2-8) shall meet engine manufacturer's approval. Consult Honeywell HTS900-2-1D Light Maintenance Manual for use of oil brands not listed herein.

Refer to the FMS-E407-789-1 for engine oil limitations.

2.4.B Engine Oil System Servicing

Capacity: 6.0 U.S. quarts (5.7 L).

Engine oil tank is located under aft fairing, and access doors are provided for filling and draining oil tank. A sight glass and filler cap dip stick are provided to determine quantity of oil in tank.

NOTE

If helicopter engine has been shut down for more than 15 minutes, scavenge oil could have drained into gearbox. Dry motor run engine for 30 seconds before checking oil level. If not accomplished, a false high engine consumption rate indication or overfilling of oil tank could result. Do not overfill engine oil tank.

Refer to Honeywell HTS900-2-1D Light Maintenance Manual for servicing instructions and oil filter change procedures.

2.4.C Transmission and Tail Rotor Gearbox Oils

Oils conforming to following specifications are approved for use in transmission and tail rotor gearbox (Table 2-10):

SPECIFICATION	OAT RANGE
DOD-PRF-85734	OAT above -40°C (-40°F)
MIL-PRF-7808 (NATO 0-148)	Below -18°C (0°F)

NOTE

It is recommended that DOD-PRF-85734 oil be used in transmission and tail rotor gearbox to maximum extent allowed by temperature limitations. Refer to the FMS-E407-789-1 for transmission and tail rotor gearbox oil limitations.

2.4.D Transmission and Tail Rotor Gearbox Servicing

Sight glasses are provided to determine quantity of oil in transmission and tail rotor gearbox.

NOTE

When checking oil levels, consider the slope of helicopter landing surface. If not considered, a false oil quantity indication or overfilling of gearbox could result.

Transmission capacity	5.0 U.S. quarts (4.7 L)
Tail rotor gearbox capacity	0.33 U.S. quarts (0.31 L)

NOTE

DOD-PRF-85734 oil is not approved for use in ambient temperatures below -40°C (-40°F). When changing to an oil of a different specification, system shall be drained and flushed.

When adding oil to the main transmission or tail rotor gearbox, the identical brand and specification of oil already in each gearbox shall be used. However, in circumstances where emergency top-off or inadvertent mixing may occur, it is acceptable to use oil with a different brand name within the same specification. No further action will be required until the next scheduled oil change, provided there is no indication of foggy or hazy oil appearance in the sight gauge.

If oils of different specifications have been mixed or a foggy or hazy oil appearance exists, accomplish the required steps per paragraph 2.4.D.1.
Refer to the ICA-E407-789, Chapter 12 for detailed procedures for draining oil and changing filters.

2.4.D.1 Oil Change – Different Specification

When changing to an oil of different specification, accomplish the following steps:

NOTE

Refer to the ICA-E407-789, Chapter 12 for the maintenance instructions.

1. Drain transmission and tail rotor gearbox.
2. Replace transmission oil filter.
3. Service transmission with proper amount of approved oil.
4. Service tail rotor gearbox with proper amount of approved oil.
5. Operate helicopter for not less than 30 minutes nor longer than 5 hours.
6. Drain transmission and tail rotor gearbox.
7. Service transmission with proper amount of approved oil.
8. Service tail rotor gearbox with proper amount of approved oil.
9. During first 100 hours of operation with new oil, check oil sight glasses closely for indications of foggy or hazy appearance. If these indications occur, repeat step 6 through step 9 until eliminated.

2.5 Hydraulic Fluid

Hydraulic fluids listed in Table 2-11 conform to MIL-PRF-5606 (NATO H-515) and are approved for use in hydraulic flight control system and rotor brake.

2.5.A Hydraulic System Servicing

Reservoir capacity 1.0 U.S. pint (0.5 L)

Hydraulic reservoir is located on top of fuselage, forward of transmission, and under forward fairing. A sight glass is provided to determine quantity of hydraulic fluid in reservoir.

Service hydraulic system as follows:

1. Open and support top of forward fairing.
2. Remove cap and fill reservoir until sight glass is full of hydraulic fluid.
3. Secure cap and fairing.

Table 2-9. Honeywell-Approved Primary Fuels

FUEL SPECIFICATION	GRADE	COUNTRY OR ORGANIZATION
Commercial Kerosene Fuels		
EMS53111	Jet A	Honeywell
ASTM D 1655	Jet A	U.S. Commercial
CAN/CGSB 3.23	Jet A	Canada
Guidance Material, 5th Edition, January 2004	Jet A	International Air Transport Association (IATA)
Low-Freeze-Point Kerosene Fuels		
EMS53112	Jet A-1	Honeywell
ASTM D 1655	Jet A-1	U.S. Commercial
DEF STAN 91-91 (DERD 2494)	Jet A-1	United Kingdom
CAN/CGSB 3.23-2005	Jet A-1	Canada
Guidance Material, 5th Edition, January 2004	Jet A-1	IATA
JSFC Issue 20, September 2005	Jet A-1	Joint Fueling System Checklist (JFSC)
MIL-DTL-83133	F-35	U.S. Military
DCSEA 134/B (AIR 3405)	F-35	France
NATO F-35	F-35	NATO Code
GOST R 52050-2003	Jet A-1	Russia
PRC GB6537-94	No. 3 Jet Fuel	China (PRC)
Low-Freeze-Point Kerosene Fuels With Fuel System Icing Inhibitor		
EMS53112	JP-8	Honeywell
MIL-DTL-83133	JP-8 (F-34)	U.S. Military
DEF STAN 91-87 (DERD 2453)	F-34	United Kingdom

FUEL SPECIFICATION	GRADE	COUNTRY OR ORGANIZATION
CAN/CGSB 3.24	F-34	Canada
DCSEA 134/B (AIR 3405)	F-34	France
NATO F-34	F-34	NATO Code
Low-Freeze-Point Kerosene Fuels With Fuel System Icing Inhibitor and Thermal Stability Improver Additive		
MIL-DTL-83133	JP-8+100	U.S. Military
NATO F-37	F-37	NATO Code
High-Flash-Point Kerosene Fuels With Fuel System Icing Inhibitor		
EMS53116	JP-5	Honeywell
MIL-DTL-5624	JP-5	U.S. Military
DEF STAN 91-86 (DERD 2452)	F-44	United Kingdom
CAN/CGSB-3.24	F-44	Canada
CAN 3-GP-24D	High Flash	Canada
DCSEA 144/B (AIR 3404)	F-44	France
NATO F-44	F-44	NATO Code
High-Flash-Point Kerosene Fuels		
DERD 2498	F-43	United Kingdom
DCSEA 144/B (AIR 3404)	XF-43	France
NATO F-43	F-43	NATO Code
Low-Flash-Point, Low-Freeze-Point Kerosene Fuels		
Guidance Material, 5th Edition, January 2004	TS-1	IATA
GOST10227-86	TS-1	Russia
GSTU 320.00149943.011-99	TS-1	Ukraine
GOST10227-86	RT	Russia
GSTU 320.00149943.007-99	RT	Ukraine
Wide-Cut Fuels		

FUEL SPECIFICATION	GRADE	COUNTRY OR ORGANIZATION
EMS 53113	Jet B	Honeywell
ASTM D 6615	Jet B	U.S. Commercial
DERD 2486	Jet B	United Kingdom
CAN/CGSB-3.22-2002	Jet B	Canada
GOST10227-86	T-2	Russia
Wide-Cut Fuels With Fuel System Icing Inhibitor		
EMS 53113	JP-4	Honeywell
MIL-DTL-5624	JP-4	U.S. Military
DEF STAN 91-88 (DERD 2454)	F-40	United Kingdom
CAN/CGSB-3.22	F-40	Canada
AIR 3407	F-40	France
NATO F-40	F-40	NATO Code

Table 2-10. Transmission and Tail Rotor Gearbox Oils

VENDOR	PRODUCT NAME
SPECIFICATION MIL-PRF-7808 (NATO O-148) (FOR OAT BELOW -18°C/0°F)	
Air BP	BO Turbo Oil 2389 BO Turbo Oil 2391
Burmah-Castrol (UK) Ltd.	Castrol 399
Castrol	Brayco 880 Castrol 399
Hatco Chemical	Hatcol 1278 Hatcol 1280
Hexagon Enterprises	Metrex AF Oil 01, 02, 07
Huls America	AO Syn Jet III PQ Turbine Oil 4236 PQ Turbine Oil 4706 PQ Turbine Oil 4707 PQ Turbine Oil 8365 PQ Turbine Oil 9900
Mobil Oil	RM-248A RM-272A
NYCO, S.A.	Turbonycoil 160
Royal Lubricants	Royco 808
Shell International	Aero Shell Turbine Oil 308
SPECIFICATION DOD-PRF85734 (FOR OAT ABOVE -40°C/-40°F)	
Air BP	BP Turbo Oil 25
Royal Lubricants	Royco Turbine Oil 555
Shell International	Aeroshell Turbine Oil 555

Table 2-11. Hydraulic Fluids – MIL-PRF-5606 (NATO H-515)

VENDOR	PRODUCT NAME
Arpol Petroleum	Arpolair 5606
Castrol	Brayco Micronic 756
Castrol-Canada	Castrol Aero HF 515
Chevron USA	Chevron Aviation Hydraulic Fluid E (PED 5597) Chevron PED 6062 Chevron PED 6063 Chevron PED 6064 Chevron PED 6065
Convoy Oil	Convoy 606
Esso SAF	Esso Fluid Aviation Invarol FJ13
Hexagon Enterprises	Metrex Hydrol 1
Huls America	PQ 4140 PQ 9300 PQ 9301 PQ 9302 PQ 9309 PQ 9310 PQ 9311
Mobil Oil	Mobil Aero HFE
NYCO S.A.	NYCO Hydraunycoil FH51
Rohm & Haas	PA 4394
Royal Lubricants	Royco 756
Shell International	AeroShell 41
Technolube	Technolube FB003

Section 3

Conversion Tables

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Section 3

Conversion Tables

3.1 Introduction

This section contains additional information which may be useful for operational planning but which is not required for inclusion in the Rotorcraft Flight Manual. Additional data may be developed and included as appropriate.

3.2 Conversion Tables

The conversion tables (Table 3-1 through Table 3-8) provide useful information to assist in flight planning and operations.

Table 3 - 1. Celsius to Fahrenheit Conversion

Celsius to Fahrenheit Conversion Table
°F = (°C x 1.8) + 32°
°C = (°F – 32°) x .555

°C → °F		°C ← °F	°C → °F		°C ← °F	°C → °F		°C ← °F
-62.2	-80	-112.0	37.8	100	212.0	126.7	260	500.0
-56.7	-70	-94.0	40.6	105	221.0	132.2	270	518.0
-51.1	-60	-76.0	43.3	110	230.0	137.8	280	536.0
-45.6	-50	-58.0	46.1	115	239.0	143.3	290	554.0
-40.0	-40	-40.0	48.9	120	248.0	148.9	300	572.0
-34.4	-30	-22.0	51.7	125	257.0	154.4	310	590.0
-31.7	-25	-13.0	54.4	130	266.0	160.0	320	608.0
-28.9	-20	-4.0	57.2	135	275.0	165.6	330	626.0
-26.1	-15	5.0	60.0	140	284.0	171.1	340	644.0
-23.3	-10	14.0	62.8	145	293.0	176.7	350	662.0
-20.6	-5	23.0	65.6	150	302.0	182.2	360	680.0
-17.8	0	32.0	68.3	155	311.0	187.8	370	698.0
-15.0	5	41.0	71.1	160	320.0	193.3	380	716.0
-12.2	10	50.0	73.9	165	329.0	198.9	390	734.0
-9.4	15	59.0	76.7	170	338.0	204.4	400	752.0
-6.7	20	68.0	79.4	175	347.0	210.0	410	770.0
-3.9	25	77.0	82.2	180	356.0	215.6	420	788.0
-1.1	30	86.0	85.0	185	365.0	221.1	430	806.0
1.6	35	95.0	87.8	190	374.0	226.7	440	824.0
4.4	40	104.0	90.6	195	383.0	232.2	450	842.0
7.2	45	113.0	93.3	200	392.0	237.8	460	860.0
10.0	50	122.0	96.1	205	401.0	243.3	470	878.0
12.8	55	131.0	98.9	210	410.0	248.9	480	896.0
15.6	60	140.0	101.7	215	419.0	254.4	490	914.0
18.3	65	149.0	104.4	220	428.0	260.0	500	932.0
21.1	70	158.0	107.2	225	437.0	265.6	510	950.0
23.9	75	167.0	110.0	230	446.0	271.1	520	968.0
26.7	80	176.0	112.8	235	455.0	276.7	530	986.0
29.4	85	185.0	115.6	240	464.0	282.2	540	1004.0
32.2	90	194.0	118.3	245	473.0	287.8	550	1022.0
35.0	95	203.0	121.1	250	482.0	293.3	560	1040.0
298.9	570	1058.0	415.6	780	1436.0	704.4	1300	2372.0
304.4	580	1076.0	426.7	800	1472.0	732.2	1350	2462.0
310.0	590	1094.0	437.8	820	1508.0	760.0	1400	2552.0
315.6	600	1112.0	454.4	850	1562.0	787.7	1450	2642.0
326.7	620	1148.0	482.2	900	1652.0	815.5	1500	2732.0
337.8	640	1184.0	510.0	950	1742.0	843.3	1550	2822.0
348.9	660	1220.0	537.7	1000	1832.0	871.1	1600	2912.0
360.0	680	1256.0	565.5	1050	1922.0	898.8	1650	3002.0
371.1	700	1292.0	593.3	1100	2012.0	926.6	1700	3092.0
382.2	720	1328.0	621.1	1150	2102.0	954.4	1750	3182.0
393.3	740	1364.0	648.8	1200	2192.0	982.2	1800	3272.0
404.4	760	1400.0	676.6	1250	2282.0	1010.0	1850	3362.0

Table 3 - 2. Gallons to Liters Conversion

Gallons To Liters Conversion Table					
US Gallon	**Imperial Gallon**	**Liter**	**US Gallon**	**Imperial Gallon**	**Liter**
10	8.33	37.85	170	141.55	643.45
20	16.65	75.71	180	149.86	681.30
30	24.98	113.56	190	158.20	719.16
40	33.31	151.42	200	166.52	757.18
50	41.63	189.27	210	174.84	795.03
60	49.96	227.13	220	183.18	832.89
70	58.28	264.98	230	191.50	870.74
80	66.61	302.83	240	199.84	908.60
90	74.94	340.69	250	208.14	946.45
100	83.26	378.54	260	216.48	984.45
110	91.59	416.35	270	224.82	1022.16
120	99.92	454.20	280	233.14	1060.01
130	108.24	492.05	290	241.56	1097.87
140	116.57	529.90	300	249.80	1135.62
150	124.90	567.75	310	258.12	1173.47
160	133.22	605.60	320	266.44	1211.33

Table 3 - 3. Inches to Millimeters Conversion

Inches to Millimeters Conversion Table										
Inches	0	1	2	3	4	5	6	7	8	9
	mm	mm	mm	mm	mm	mm	mm	mm	mm	mm
0	–	25.4	50.8	76.2	101.6	127.0	152.4	177.8	203.2	228.6
10	254.0	279.4	304.8	330.2	355.6	381.0	406.4	431.8	457.2	482.6
20	508.0	533.4	558.8	584.2	609.6	635.0	660.4	685.8	711.2	736.6
30	762.0	787.4	812.8	838.2	863.6	889.0	914.4	939.8	965.2	990.6
40	1016.0	1041.4	1066.8	1092.2	1117.6	1143.0	1168.4	1193.8	1219.2	1244.6
50	1270.0	1295.4	1320.8	1346.2	1371.6	1397.0	1422.4	1447.8	1473.2	1498.6
60	1524.0	1549.4	1574.8	1600.2	1625.6	1651.0	1676.4	1701.8	1727.2	1752.6
70	1778.0	1803.4	1828.8	1854.2	1879.6	1905.0	1930.4	1955.8	1981.2	2006.6
80	2032.0	2057.4	2082.8	2108.2	2133.6	2159.0	2184.4	2209.8	2235.2	2260.6
90	2286.0	2311.4	2336.8	2362.2	2387.6	2413.0	2438.4	2463.8	2489.2	2514.6
100	2540.0	2565.4	2590.8	2616.2	2641.6	2667.0	2692.4	2717.8	2743.2	2768.6

Table 3 - 4. Feet to Meters Conversion

Feet to Meters Conversion Table										
Feet	0	1	2	3	4	5	6	7	8	9
	Meters	Meters	Meters	Meters	Meters	Meters	Meters	Meters	Meters	Meters
0	–	0.305	0.610	0.914	1.219	1.524	1.829	2.134	2.438	2.743
10	3.048	3.353	3.658	3.962	4.267	4.572	4.877	5.182	5.486	5.791
20	6.096	6.401	6.706	7.010	7.315	7.620	7.925	8.229	8.534	8.839
30	9.144	9.449	9.753	10.058	10.363	10.668	10.972	11.277	11.582	11.887
40	12.192	12.496	12.801	13.106	13.411	13.716	14.020	14.325	14.630	14.935
50	15.240	15.544	15.849	16.154	16.459	16.763	17.068	17.373	17.678	17.983
60	18.287	18.592	18.897	19.202	19.507	19.811	20.116	20.421	20.726	21.031
70	21.335	21.640	21.945	22.250	22.555	22.859	23.164	23.469	23.774	24.070
80	24.383	24.688	24.993	25.298	25.602	25.907	26.212	26.517	26.822	27.126
90	27.431	27.736	28.041	28.346	28.651	28.955	29.260	29.565	29.870	30.174
100	30.479	30.784	31.089	31.394	31.698	32.003	32.308	32.613	32.918	33.222

Table 3 - 5. Pounds to Kilograms Conversion

Pounds to Kilograms Conversion Table										
Pounds	0	1	2	3	4	5	6	7	8	9
	Kg	Kg	Kg	Kg	Kg	Kg	Kg	Kg	Kg	Kg
0	–	0.454	0.907	1.361	1.814	2.268	2.722	3.175	3.629	4.082
10	4.536	4.990	5.443	5.897	6.350	6.804	7.257	7.711	8.165	8.618
20	9.072	9.525	9.979	10.433	10.886	11.340	11.793	12.247	12.701	13.154
30	13.608	14.061	14.515	14.969	15.422	15.876	16.329	16.783	17.237	17.690
40	18.144	18.597	19.051	19.504	19.958	20.412	20.865	21.319	21.722	22.226
50	22.680	23.133	23.587	24.040	24.494	24.948	25.401	25.855	26.308	26.762
60	27.216	27.669	28.123	28.576	29.030	29.484	29.937	30.391	30.844	31.298
70	31.751	32.205	32.659	33.112	33.566	34.019	34.473	34.927	35.380	35.834
80	36.287	36.741	37.195	37.648	38.102	38.555	39.009	39.463	39.916	40.370
90	40.823	41.277	41.730	42.184	42.638	43.091	43.545	43.998	44.453	44.906
100	45.359	45.813	46.266	46.720	47.174	47.627	48.081	48.534	48.988	49.442

Table 3 - 6. Velocity Conversion

Velocity Conversion Table							
Knots	MPH	Km/HR	Meters/sec	Knots	MPH	Km/HR	Meters/sec
5	5.8	9.3	2.6	105	120.8	194.4	54.0
10	11.5	18.5	5.1	110	126.6	203.7	56.6
15	17.3	27.8	7.7	115	132.3	213.0	59.2
20	23.0	37.0	10.3	120	138.1	222.2	61.7
25	28.8	46.3	12.9	125	143.8	231.5	64.3
30	34.5	55.6	15.4	130	149.6	240.7	66.9
35	40.3	64.8	18.0	135	155.4	250.0	69.4
40	46.0	74.1	20.6	140	161.1	259.3	72.0
45	51.8	83.3	23.1	145	166.9	268.5	74.6
50	57.5	92.6	25.7	150	172.6	277.8	77.2
55	63.3	101.9	28.3	155	178.4	287.0	79.7
60	69.0	111.1	30.9	160	184.1	296.3	82.3
65	74.8	120.4	33.4	165	189.9	305.6	84.9
70	80.6	129.6	36.0	170	195.6	314.8	87.4
75	86.3	138.9	38.6	175	201.4	324.1	90.0
80	92.1	148.1	41.2	180	207.1	333.3	92.6
85	97.8	157.4	43.7	185	212.9	342.6	95.2
90	103.6	166.7	46.3	190	218.6	351.9	97.8
95	109.3	175.9	48.9	195	224.4	361.1	100.3
100	115.1	185.2	51.4	200	230.2	370.4	102.9

Table 3 - 7. Standard Atmosphere

Standard Atmosphere Table

Standard Conditions:

Temperature	15°C (59°F)
Pressure	29.921 IN. Hg (2116.216 Lb/sq ft.)
Density	0.0023769 Slugs/Cu. Ft
Speed of Sound	1116.89 Ft/sec (661.7 Knots)

Conversion Factors:

1 IN. Hg =70.727 Lb/sq ft
1 IN. Hg =0.49116 Lb/sq IN.
1 Knot = 1.151 M.P.H.
1 Knot = 1.688 ft/sec

Altitude Feet	Density Ratio σ	1/√σ	Temp °C	Temp °F	Speed of Sound Knots	Pressure IN. Hg	Pressure Ratio
0	1.0000	1.0000	15.000	59.000	661.7	29.921	1.0000
1000	0.9711	1.0148	13.019	55.434	659.5	28.856	0.9644
2000	0.9428	1.0299	11.038	51.868	657.2	27.821	0.9298
3000	0.9151	1.0454	9.056	48.302	654.9	26.817	0.8962
4000	0.8881	1.0611	7.076	44.735	652.6	25.842	0.8637
5000	0.8617	1.0773	5.094	41.169	650.3	24.896	0.8320
6000	0.8359	1.0938	3.113	37.603	648.7	23.978	0.8014
7000	0.8106	1.1107	1.132	34.037	645.6	23.088	0.7716
8000	0.7860	1.1279	-0.850	30.471	643.3	22.225	0.7428
9000	0.7620	1.1456	-2.831	26.905	640.9	21.388	0.7148
10,000	0.7385	1.1637	-4.812	23.338	638.6	20.577	0.6877
11,000	0.7155	1.1822	-6.793	19.772	636.2	19.791	0.6614
12,000	0.6932	1.2011	-8.774	16.206	633.9	19.029	0.6360
13,000	0.6713	1.2205	-10.756	12.640	631.5	18.292	0.6113
14,000	0.6500	1.2403	-12.737	9.074	629.0	17.577	0.5875
15,000	0.6292	1.2606	-14.718	5.508	626.6	16.886	0.5643
16,000	0.6090	1.2815	-16.699	1.941	624.2	16.216	0.5420
17,000	0.5892	1.3028	-18.680	-1.625	621.8	15.569	0.5203
18,000	0.5699	1.3246	-20.662	-5.191	619.4	14.942	0.4994
19,000	0.5511	1.3470	-22.643	-8.757	617.0	14.336	0.4791
20,000	0.5328	1.3700	-24.624	-12.323	614.6	13.750	0.4595
21,000	0.5150	1.3935	-26.605	-15.899	612.1	13.184	0.4406
22,000	0.4976	1.4176	-28.587	-19.456	609.6	12.636	0.4223
23,000	0.4806	1.4424	-30.568	-23.022	607.1	12.107	0.4046
24,000	0.4642	1.4678	-32.549	-26.588	604.6	11.597	0.3874
25,000	0.4481	1.4938	-34.530	-30.154	602.1	11.103	0.3711

Table 3 - 8. Barometric Pressure Conversion

Inches to Millibars										
Mercury Inches	**0.00**	**0.01**	**0.02**	**0.03**	**0.04**	**0.05**	**0.06**	**0.07**	**0.08**	**0.09**
	Millibars									
28.0−	948.2	948.5	948.9	949.2	949.5	949.9	950.2	950.6	950.9	951.2
28.1−	951.6	951.9	952.3	952.6	952.9	953.3	953.6	953.9	954.3	954.6
28.2−	955.0	955.3	955.6	956.0	956.3	956.7	957.0	957.3	957.7	958.0
28.3−	958.3	958.7	959.0	959.4	959.7	960.0	960.4	960.7	961.1	961.4
28.4−	961.7	962.1	962.4	962.8	963.1	963.4	963.8	964.1	964.4	964.8
28.5−	965.1	965.5	965.8	966.1	966.5	966.8	967.2	967.5	967.8	968.2
28.6−	968.5	968.8	969.2	969.5	969.9	970.2	970.5	970.9	971.2	971.6
28.7−	971.9	972.2	972.6	972.9	973.2	973.6	973.9	974.3	974.6	974.9
28.8−	975.3	975.6	976.0	976.3	976.6	977.0	977.3	977.7	978.0	978.3
28.9−	978.7	979.0	979.3	979.7	980.0	980.4	980.7	981.0	981.4	981.7
29.0−	982.1	982.4	982.7	983.1	983.4	983.7	984.1	984.4	984.8	985.1
29.1−	985.4	985.8	986.1	986.5	987.8	987.1	987.5	987.8	988.2	988.5
29.2−	988.8	989.2	989.5	989.8	990.2	990.5	990.9	991.2	991.5	991.9
29.3−	992.2	992.6	992.9	993.2	993.6	993.9	994.2	994.6	994.9	995.3
29.4−	995.6	995.9	996.3	996.6	997.0	997.3	997.6	998.0	998.3	998.6
29.5−	999.0	999.3	999.7	1000.0	1000.4	1000.7	1001.0	1001.4	1001.7	1002.0
29.6−	1002.4	1002.7	1003.1	1003.4	1003.7	1004.1	1004.4	1004.7	1005.1	1005.4
29.7−	1005.8	1006.1	1006.4	1006.8	1007.1	1007.5	1007.8	1008.1	1008.5	1008.8
29.8−	1009.1	1009.5	1009.8	1010.2	1010.5	1010.8	1011.2	1011.5	1011.9	1012.2
29.9−	1012.5	1012.9	1013.2	1013.5	1013.9	1014.2	1014.6	1014.9	1015.2	1015.6
30.0−	1015.9	1016.3	1016.6	1016.9	1017.3	1017.6	1018.0	1018.3	1018.6	1019.0
30.1−	1019.3	1019.6	1020.0	1020.3	1020.7	1021.0	1021.3	1021.7	1022.0	1022.4
30.2−	1022.7	1023.0	1023.4	1023.7	1024.0	1024.4	1024.7	1025.1	1025.4	1025.7
30.3−	1026.1	1026.4	1026.7	1027.1	1027.4	1027.8	1028.1	1028.4	1028.8	1029.1
30.4−	1029.5	1029.8	1030.1	1030.5	1030.8	1031.2	1031.5	1031.8	1032.2	1032.5
30.5−	1032.9	1033.2	1033.5	1033.9	1034.2	1034.5	1034.9	1035.2	1035.5	1035.9
30.6−	1036.2	1036.6	1036.9	1037.3	1037.6	1037.9	1038.3	1038.6	1038.9	1039.3
30.7−	1039.6	1040.0	1040.3	1040.6	1041.0	1041.3	1041.7	1042.0	1042.3	1042.7
30.8−	1043.0	1043.3	1043.7	1044.0	1044.4	1044.7	1045.0	1045.4	1045.7	1061.1
30.9−	1046.4	1046.7	1047.1	1047.4	1047.8	1048.1	1048.4	1048.8	1049.1	1049.5
Millibars to Inches										
Millibars	**0**	**1**	**2**	**2**	**4**	**5**	**6**	**7**	**8**	**9**
	Inches									
940	27.76	27.79	27.82	27.85	27.88	27.91	27.94	27.96	27.99	28.02
950	28.05	28.08	28.11	28.14	28.17	28.20	28.23	28.26	28.29	28.32
960	28.35	28.38	28.41	28.44	28.47	28.50	28.53	28.56	28.58	28.61
970	28.64	28.67	28.70	28.73	28.76	28.79	28.82	28.85	28.88	28.91
980	28.94	28.97	29.00	29.03	29.06	29.09	29.12	29.15	29.18	29.21
990	29.23	29.26	29.29	29.32	29.35	29.38	29.41	29.44	29.47	29.50
1000	29.53	29.56	29.59	29.62	29.65	29.68	29.71	29.74	29.77	29.80
1010	29.83	29.85	29.88	29.91	29.94	29.97	30.00	30.03	30.06	30.09
1020	30.12	30.15	30.18	30.21	30.24	30.27	30.30	30.33	30.36	30.39
1030	30.42	30.45	30.47	30.50	30.53	30.56	30.59	30.62	30.65	30.68
1040	30.71	30.74	30.77	30.80	30.83	30.86	30.89	30.92	30.95	30.98
1050	31.01	31.04	31.07	31.09	31.12	31.15	31.18	31.21	31.24	31.27